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AIRWORTHINESS AND FLIGHT CHARACTERISTICS (A&FC) TEST OF YAH-64 ADVANCED ATTACK HELICOPTER, PROTOTYPE QUALIFICATION TEST-GOVERNMENT (PQT-G), PART 3 AND PRODUCTION VALIDATION TEST-GOVERNMENT (PVT-G) FOR HANDBOOK VERIFICATION

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OCTOBER 1982

FINAL REPORT

Approved for public release, distribution unlimited

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The Airworthiness and Flight Characteristics Test of the YAH-64 helicopter (USA S/N 77-23258) was conducted between 12 May and 19 August 1982. Seventy flights and 70.4 productive hours were flown. Performance testing assessed the probability of meeting the production contract requirements and consisted of an evaluation of hover, takeoff, level flight, forward flight climb and autorotational descent performance. Handling qualities testing determined compliance with selected research and development contract requirements and

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included standard stability and control tests, an evaluation of slope landing characteristics and instrument flight capability. Additional tests included evaluation of an uprated engine and an external noise survey. The YT700-GE-701 engines provided a significant increase in power available, compared to the YT700-GE-700R engines, and proved very reliable. Hover, verticl climb and level flight performance was significantly improved by the YT700-GE-701 engines. The YAH-64 now meets the performance requirements of the system specification for the production program; the vertical climb and maximum level flight cruise speeds. Slope landing characteristics were satisfactory up to 9 degree lateral slopes and 10 degree longitudinal slopes. The automatic contingency power feature of the YT700-GE-701 engines is an enhancing characteristic. One deficiency, the possibility of a false indication of a dual engine failure in the event of a single engine failure, has not been adequately corrected. Five previously reported shortcomings have been corrected. Three additional shortcomings were identified.

YAH-64 "Apache" Advanced Attack Helicopter

DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
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DRDAV-D

SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 80-17-3, Airworthiness and Flight Characteristics (A&FC) Test of YAH-64 Advanced Attack Helicopter, Prototype Qualification Test - Government (PQT-G), Part 3 and Production Validation Test - Government (PVT-G) for Handbook Verification

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1. The purpose of this letter is to establish the Directorate for Development and Qualification position on the subject report. The objectives of this evaluation were to assess the performance and handling qualities of the YAH-64 incorporating YT700-GE-701 engines. Performance testing was conducted to assess the probability of meeting the production contract requirements and to provide data for preparation of the operator's manual. Handling qualities testing was accomplished to determine compliance with selected research and development contract requirements. Due to schedule restraints, the A&FC test program was divided into three parts, of which this is the third.

2. This Directorate agrees with the report conclusions and recommendations, with the exceptions identified herein. Dispositions of redesigned subsystems/components affecting the conclusions are also identified. Conclusions are discussed by paragraphs, as indicated.

a. Paragraph 73c. The limited slope landing evaluation conducted during this A&FC has shown that large fuselage roll attitudes can be reached. The narrow main landing gear width (6.66 feet) designed to accommodate transportability requirements is the major factor contributing to the high roll attitudes achieved. Other factors include assymetric loadings, improper strut servicing, and improper tire inflation. This test was conducted under controlled conditions where the slope angles had been measured, struts and tires serviced, and test instrumentation installed. This would not be the case in operational use. In the interest of flight safety the AH-64 will be limited to landings on lateral slopes up to 8° and fuselage roll attitude should be limited to 10° as determined by the attitude indicator. Additionally, successful aircraft shutdown/startup must be demonstrated under these conditions. The Program Manager has requested the user to rewrite this requirement. The user is in the process of amending his requirements for slope landings to 8° lateral and 10° longitudinal. One of the governing factors is

the capability of ground support vehicles to operate on these slopes, which they cannot accomplish at current specification values. On receipt of the approved material need (MN) changes, the AH-64 specification will be changed.

b. Paragraph 75. The power turbine speed (Np) engine-out warning threshold was reduced from 89% to 86% for this test. This change was not successful in correcting this previously reported problem. Hughes Helicopters, Inc. (HHI) is currently evaluating a modification to prevent a false indication of a dual engine failure by inhibiting the Np warning as long as the engine is producing significant torque. This system would require both Np and torque to be low in order to activate the engine-out warning system. Such a system should provide a timely warning of an engine failure without the possibility of a false indication of a dual engine failure. The overall situation is considered a questionable deficiency because there are no significant differences in what the pilot's immediate actions should be in this situation.

c. Paragraph 76a. The incorporation of a collective anticipator system is under consideration to minimize main rotor speed droop during power application from a zero torque condition. General Electric is currently evaluating the data to determine what action may be taken to correct the engine/airframe oscillation noted.

d. Paragraph 76e. The Environmental Control System has been redesigned to incorporate a revised flow distribution and airframe sealing. These changes have been incorporated, tested, and should alleviate the shortcoming.

e. Paragraph 76k. The production configuration incorporates thermal lockout switch improvements in the filter assemblies which should eliminate the faulty activation noted during this test. Additionally, the production configuration will provide assemblies with twice the effective filter area, to decrease the pressure drop across the filter, and an improved filter bypass button system.

f. The position of this Directorate concerning the remaining previously reported shortcomings is specified in EDT-4 and A&FC, Part 1, and remains unchanged.

3. This Directorate concurs with the recommendations presented, with the exception of paragraphs 78f and 78g. The Directorate position concerning these recommendations is specified in paragraph 2a of this letter.

4. The installation of the YT700-GE-701 engines provided a significant increase in power available compared to the TY700-GE-700R engines used during previous evaluations. Significant improvements were noted in hover and

DRDAV-D
SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 80-17-3, Airworthiness and Flight Characteristics (A&FC) Test of YAH-64 Advanced Attack Helicopter, Prototype Qualification Test - Government (PQT-G), Part 3 and Production Validation Test - Government (PVT-G) for Handbook Verification

vertical climb performance and in maximum level flight airspeed. These improvements have resulted in the AH-64 flight performance requirements now being achievable.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Director of Development
and Qualification

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INTRODUCTION

BACKGROUND

1. In June 1973, the United States Army Aviation Systems Command, since renamed the US Army Aviation Research and Development Command (AVRADCOM), awarded a Phase 1 Advanced Development Contract to Hughes Helicopters Incorporated (HHI). The contract required HHI to design, develop, fabricate, and initiate a development/qualification effort on two Advanced Attack Helicopter prototypes and a ground test vehicle as part of a Government Competitive Test. The United States Army Aviation Engineering Flight Activity (USAAEFA) conducted Development Test I (ref 1, app A) using two of these aircraft. In December 1976, AVRADCOM awarded a Phase 2 Engineering Development Contract to HHI for engineering development. Engineer Design Tests (EDT) 1 and 2 (refs 2 and 3) were conducted by USAAEFA to evaluate development progress. In December 1980, USAAEFA conducted EDT 4 (ref 4) to evaluate the performance and flight handling characteristics of a new empennage configuration. In December 1980 and January 1981 USAAEFA, in conjunction with the US Army Aviation Development Test Activity, conducted EDT-5 (ref 5) to assess the integrated operation of all YAH-64 subsystems. AVRADCOM requested USAAEFA to conduct an Airworthiness and Flight Characteristics (A&FC) test following incorporation of as many production changes as practical. Part 1 of this A&FC test (ref 6) was completed on 17 July 1981 and Part 2 (ref 7) was completed on 17 December 1981. USAAEFA was tasked to conduct the remainder of this A&FC during May through July 1982 (ref 8). A test plan (ref 9) was submitted in January 1982 and an airworthiness release (ref 10) was issued in April 1982.

TEST OBJECTIVES

2. The objective of this A&FC was to complete the Prototype Qualification Test-Government (POT-G), Part 3 and to conduct a Production Validation Test-Government (PVT-G). The specific objectives of each test were as follows:

a. The objective of POT-G was to evaluate the YAH-64 with the YT700-GE-701 engine relative to handling qualities, slope landings, and an external acoustical noise survey.

b. The objective of the PVT-G was to obtain hover, takeoff, forward flight climb, level flight and autorotational descent performance data of the YAH-64 with the YT700-GE-701 engine for operator's manual preparation.

DESCRIPTION

3. The YAH-64 (USA S/N 77-23258) is a two-place, tandem seat, twin engine helicopter with four-bladed main and antitorque rotors and conventional wheel landing gear. The helicopter is powered by two General Electric (GE) YT700-GE-701 turboshaft engines which replace the YT700-GE-700R engines used on previous evaluations. The YAH-64 has a movable horizontal stabilator with three modes of operation: Manual, Automatic and Nap-of-the-earth (NOE)/Approach. A 30mm gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has a wing with two store pylons on each side for carrying HELLFIRE missiles, 2.75-inch (in.) folding fin aerial rockets or external fuel tanks. Further description of the helicopter may be found in the system specification (ref 11, app A), the operator's manual (ref 12), and appendix B.

TEST SCOPE

4. Flight testing was conducted at Yuma Proving Ground, Arizona (elevation 800 feet), and Bishop, California (test site elevations of 4120 feet and 9980 feet). Tests were conducted during the period 13 May 1982 through 19 August 1982. Seventy test flights were conducted for a total of 70.4 productive flight hours. HHI installed, calibrated and maintained the test instrumentation and performed all aircraft maintenance during the test. Flight restrictions contained in the airworthiness release issued by AVRADCOM and the operator's manual were observed during this evaluation. Performance testing was conducted to assess the probability of meeting the production contract requirements. Handling qualities and other tests were conducted to determine compliance with selected research and development contract requirements. Specific test conditions are presented in the Results and Discussion section of this report. All tests were flown by a USAAEFA crew consisting of two test pilots. Where possible, flight test data were compared with the system specification for the AH-64 production program (ref 11, app A) and results obtained during previous evaluations.

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures (refs 13 and 14, app A) were used during this evaluation. Test methods are briefly discussed in the Results and Discussion section of this report. Flight test data were obtained from calibrated test instrumentation and were recorded on magnetic

tape. Real time telemetry was used to monitor selected parameters throughout the flight test program. A detailed listing of the test instrumentation is contained in appendix C. Test techniques and data analysis methods are described in appendix D. The Handling Qualities Rating Scale (HQRS) and the Vibration Rating Scale (VRS), shown in appendix D, were used to quantify pilot comments.

RESULTS AND DISCUSSION

GENERAL

6. Performance and handling qualities tests were accomplished during this test program to evaluate the YAH-64 with the YT700-GE-701 engine installation. Performance testing was conducted under the production contract requirements and consisted of an evaluation of hover, takeoff, level flight, forward flight climb and autorotational descent performance. Handling qualities testing was accomplished under the research and development contract requirements and included standard stability and control tests, an evaluation of slope landing characteristics and an evaluation of instrument flight capability. Additional tests included an engine evaluation and an external noise survey. Test conditions are shown in tables 1 and 2. The YT700-GE-701 engines provided a significant increase in power available, as compared to the YT700-GE-700R engines, and proved to be very reliable throughout the test program. The hover, vertical climb and level flight performance was significantly improved by the YT700-GE-701 engines. The AH-64A now meets two of the performance requirements of the system specification for the production program (ref [illegible], app A); the vertical climb and maximum level flight cruise speeds. Slope landing characteristics were satisfactory up to 9 degree lateral slopes and 10 degree longitudinal slopes. The automatic contingency power feature of the YT700-GE-701 engines is an enhancing characteristic. One previously reported deficiency, the possibility of a false indication of a dual engine failure in the event of a single engine failure, still remains. Two previously reported shortcomings have been corrected, however, three additional shortcomings were identified.

PERFORMANCE

General

7. Performance flight testing was conducted at test site elevations of [illegible], [illegible], and 9980 feet. Tests included hover, takeoff, forward flight climb, level flight and autorotational descent performance. Test results from Parts 1 and 2 of the overall A&FC test are included, as appropriate, since many tests were conducted to extend the range of the previous data. Some minor refinements to the faired curves through the previous hover and level flight data were made. Vertical climb performance calculations were also revised. Power available and fuel flow characteristics for the YT700-GE-701 engine were provided by AVRADCOM (ref 15, app A). This data assumes no difference between the left and right engine and was derived from the GE computer program deck number 81067, dated 4 May 1982, using installation

Table 1. Test Conditions[1]
Performance Tests

Type of Test	Average/Range Gross Weight (lb)	Average Longitudinal cg (FS)[2]	Average/Range Density Altitude (ft)	Trim Calibrated Airspeed Range (kt)	Wing Stores	Remarks
Hover[3] Performance	14,660	205.7 (AFT)	5260	zero	No wing stores or pylons	11 and 21 foot wheel heights
	14,020	205.7 (AFT)	10,800			5, 11, 21 and 100 foot wheel heights
Takeoff Performance	14,640 to 16,640	202.6 (FWD)	10,960	20-55	8-HELLFIRE/ 2.75 in. rocket pods[4]	Level acceleration technique, 5 ft main wheel height
Forward Flight Climb Performance	15,000 to 16,540	204.1	3580 to 14,780	65 to 75	8-HELLFIRE	K_W[5] climbs
	15,500	203.4	4420 to 14,300	65 to 74		K_P[6] climbs
Level Flight Performance	14,820 to 15,500	202.0 (FWD)	3240 to 15,880	43 to 152	8-HELLFIRE	Constant gross weight over density ratio, constant rotor speed method
	15,780 to 16,620	201.8 (FWD)	7300 to 13,260	39 to 130	16-HELLFIRE	
	14,200	203.8	11,640 to 16,260	39 to 156	Clean	
	13,860	206.9 (AFT)	6680 to 12,760	39 to 157	Clean	
	14,660	202.2 (FWD)	11,120 to 15,560	44 to 131	2-Ferry Tanks	
	15,500	202.7 (FWD)	8200 to 10,800	81 to 132	8-HELLFIRE	Sideslip effects
Autorotational Descent Performance	15,220	204.9	5000, 10,180	43 to 121	8-HELLFIRE	Main rotor speed range: 271 to 302 RPM
	16,500	204.4 (AFT)	5080, 9940	43 to 103		

NOTES:

[1]All tests at 100% main rotor speed (289 RPM), Digital Automatic Stabilization Equipment ON and ATTITUDE HOLD OFF, except as noted. Lateral cg at buttline 0.8 left.
[2]FS = Fuselage Station. Maximum forward cg limit of FS 201 as was unattainable due to aircraft ballast limitations.
[3]Main rotor speed 281 to 298 RPM (97 to 103 percent)
[4]Two 19-round 2.75-in. rocket pods installed on outboard wing pylons to allow for ballast changes between data points.
[5]K_W = Weight correction factor
[6]K_P = Power correction factor

Table 2. Test Conditions[1]
Handling Qualities and Other Tests

Type of Test	Average Gross Weight (lb)	Average Longitudinal CG (FS)[2]	Average Density Altitude (ft)	Trim Calibrated Airspeed (kt)	Wing Stores	Remarks
Control Positions In Trimmed Forward Flight	14,940 to 15,520	202.0 (FWD) and 206.0 (AFT)	4640 to 14,920	39 to 166	8-HELLFIRE	Level flight
	15,500	206.2 (AFT)	5060	42 to 153	8-HELLFIRE Asymmetric[3]	
	14,660	202.2 (FWD)	11,120 and 15,560	42 to 131	2-Ferry Tank	
	15,200 to 16,700	204.6	4960 and 10,180	41 to 121	8-HELLFIRE	Autorotation
	15,460	202.4 (FWD)	5220	103		Climbs and autorotation
Static Longitudinal Stability	15,240	206.2 (AFT)	15,040	58, 96		±20 KIAS[4] from trim
	15,120	202.0 (FWD)	4780	58, 151	8-HELLFIRE	
	15,440	206.2 (AFT)	5000	60, 132	8-HELLFIRE Asymmetric[3]	
Static Lateral-Directional Stability	15,420	202.5 (FWD)	5140	60, 136		To sideslip limit both left and right
	14,740	206.0 (AFT)	14,760	60, 95	8-HELLFIRE	
	15,040	206.0 (AFT)	5120	60, 131	8-HELLFIRE Asymmetric[3]	
	17,680	204.1 (AFT)	5060	60, 132	16-HELLFIRE	
Maneuvering Stability	15,440	205.7 (AFT)	14,740	96		Left and right constant airspeed turns
	15,660	202.0 (FWD)	5880	139	8-HELLFIRE	
	15,560	205.9 (AFT)	6200	136	8-HELLFIRE Asymmetric[3]	
Dynamic[5] Stability	17,440	204.1 (AFT)	5120	63, 78, 100, 145	16-HELLFIRE	Longitudinal long and short period and lateral-directional oscillations
	14,860	205.4 (AFT)	6500	62, 78, 101, 148	8-HELLFIRE Asymmetric[3]	
	15,420	205.5 (AFT)	14,780	62, 80, 99		
	15,420	202.5 (FWD)	5380	60, 78, 96, 150	8-HELLFIRE	
Controllability[5]	14,880	205.6 (AFT)	14,680	78	8-HELLFIRE	Longitudinal and lateral only
Slope Landing Characteristics	14,480	204.8	2780	zero	Clean[6]	Slope angles of 4 to 12°
Power Management	14,260	205.5 (AFT)	8200	0 to 141	Clean	Engine response, functional checks, qualitative evaluation
	15,520	205.6 (AFT)	5230	97	8-HELLFIRE	
Instrument[7] Flight Capability	15,980	204.9	2760 to 6500	0 to 114	8-HELLFIRE	Simulated instrument meterological conditions
DASE[8] Evaluation	15,460	203.9	2680 to 6680	zero 46 to 142	8-HELLFIRE	Hover, low speed and forward flight
Simulated Single Engine Failures	15,520	205.6 (AFT)	2620	zero	8-HELLFIRE	Hover
			7220	49 to 138		Level flight, climb and autorotation
Vibration Characteristics	15,360	202.1 (FWD)	15,800	38 to 100	8-HELLFIRE	Level Flight
	14,700	202.3 (FWD)	15,320	37 to 94	2-Ferry Tank	
Engine[9] Starts	Note 10	Note 10	2620 and 5080	zero	Note 10	Ground starts
	14,300	204.3	17,720 (maximum)	75	Clean	Inflight starts
External Noise Survey	15,480	205.7 (AFT)	940	38 to 152		Ground based survey
	15,480	205.7 (AFT)	6520	52 to 123	8-HELLFIRE	Inflight survey

NOTES:

[1]All tests at 100% main rotor speed (289 RPM), Digital Automatic Stabilization Equipment (DASE) ON and ATTITUDE HOLD OFF except as noted. Lateral cg at buttline 0.8 left except as noted.
[2]FS = Fuselage Station. Maximum forward cg limit of FS 201 unattainable due to aircraft ballast limitations.
[3]8-HELLFIRE on left wing only simulating asymmetric jettison from 16-HELLFIRE configuration. Lateral cg at buttline 7.0 left.
[4]KIAS = Knots indicated airspeed
[5]DASE ON and OFF
[6]Clean configuration with outboard pylons installed
[7]DASE ON and OFF; ATTITUDE HOLD ON and OFF
[8]DASE ON and OFF; ATTITUDE HOLD ON and OFF, Hover Augmentation System ON and OFF
[9]Main rotor speed range 0 to 100% during ground starts
[10]Not applicable since engine starts evaluated on the ground.

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losses determined by AVRADCOM from HHI propulsion testing. Summary performance data were compared, where applicable, to the requirements of the system specification (ref 11) for the production program. The hover, vertical climb and level flight performance is significantly improved by the installation of the YT700-GE-701 engines. The YAH-64 now meets two of the performance requirements of the system specification for the production program (ref 11, app A), the vertical climb and maximum level flight cruise speeds. There are no system specifications for hover, takeoff, forward flight climb or autorotational descent performance.

Hover Performance

8. The hover performance of the YAH-64 was evaluated by determining the engine power required to hover at various wheel heights, rotor speeds and pressure altitudes. Hover tests were conducted in winds of three knots or less using the tethered and free flight hover methods at the conditions of table 1. Data at a wheel height of 100 feet were obtained previously during the A&FC Part 1 at the 4120-foot test site. Variations in thrust coefficient during tethered hover were obtained by incrementally varying rotor speed from 281 to 298 RPM (97 to 103 percent) and engine power, thus cable tension. A tensiometer was used to measure tension in the cable (photo 1). All hover tests were conducted with the aircraft in the clean configuration (no wing stores or pylons). A summary of hover performance is presented in figure 1, appendix E and nondimensional tests results are presented in figures 2 through 5.

9. The YAH-64 out-of-ground effect (OGE) hover ceiling using intermediate rated power (IRP) at the primary mission gross weight of 14,694 pounds on a standard day was 11,020 feet. At the maximum alternate gross weight of 17,650 pounds, the OGE hover ceiling was 5760 feet for standard day conditions and 1760 feet with a hot day temperature (35°C). The OGE hover capability for a hot day (35° C), 4000 feet pressure altitude was determined to be 16,180 pounds using IRP.

Takeoff Performance

10. Takeoff performance tests were conducted at the conditions of table 1 in winds 3 knots or less to determine the takeoff distance required to clear a 50-foot obstacle. The level acceleration takeoff method was used for these tests (photo 2). The aircraft was configured with 8-HELLFIRE missiles and two 19-round rocket pods to facilitate ballast changes during testing. Takeoffs were initiated from a stabilized 5-foot wheel height hover at


05

Photo 2. Level Acceleration Takeoff Performance Technique

gross weights and density altitudes where excess power was limited. Simultaneous application of forward cyclic and increased collective control were used to obtain IRP and initiate a level acceleration. At the condition where there was no excess power and the aircraft could just hover at 5 feet with IRP, acceleration was initiated by slight forward cyclic application only. The collective control was then increased to maintain IRP as the airspeed increased. Since both boom and ship system airspeed indications were unusable below 30 knots, climbout airspeeds were varied by initiating rotation at various doppler ground speeds at the command of the copilot. A constant pitch attitude, near that observed during hover, was maintained during the climbout, using the Electronic Attitude Direction Indicator (EADI) as a reference. Upon initiating acceleration from a hover, the aircraft began to climb requiring additional forward cyclic to maintain a 5-foot wheel height. Approaching effective translational lift (ETL), 15 to 20 knots true airspeed (KTAS), the aircraft began to settle slightly, but upon attaining ETL it again began to climb. All takeoffs to clear a 50-foot obstacle were successful except where climbout airspeed was below ETL. This resulted in the aircraft climbing above 50 feet initially but then descending below the 50-foot obstacle. Pilot effort to control attitude and altitude was minimal (HQRS 3). The horizontal distance required to clear and maintain clearance over a 50-foot obstacle at various climbout airspeeds is presented in figure 6, appendix E. The test data for each of the excess power conditions are presented in figures 7 through 10.

11. Tests were conducted in the primary mission configuration (para 1, app B) with winds less than 5 knots to determine the best method of applying the level acceleration takeoff technique for operational use. Since the left and right ship pitot-static systems were unusable for initiation of climb due to erroneous and unuseable indications below approximately 30 knots indicated airspeed (KIAS), a technique based on attitude reference was developed. A typical time history of a takeoff using this technique is presented as figure 11, appendix E. The most effective method was to stabilize the aircraft at a 5-foot wheel height hover (indicated radar altitude 3 to 4 feet) noting the hover pitch attitude on the EADI. A level acceleration was initiated and maintained until passing through ETL (as noted by an abrupt decrease in airframe vibration and buffet) then the aircraft was immediately rotated to the original hover pitch attitude, thus initiating a climb. This technique consistently produced a climbout airspeed of 28 to 32 KTAS, as recorded from the air data system and a doppler ground speed ranging from 57 to 62 kilometers per hour (30.8 to 33.5 knots). Attempting rotation early in the level acceleration produced a climb initially but the aircraft

did not always pass through ETL and began to descend after initially climbing through 50 feet. For this reason, the rotation should be delayed until passing through ETL to insure consistent, safe takeoffs over a 50-foot obstacle in the minimum practical distance. Due to unuseable airspeed indications, takeoff performance data for climbout airspeeds below 35 KTAS should not be presented in Chapter 7 of the operator's manual. The procedures for level acceleration takeoffs, specified in paragraph 8-50 of the operator's manual (ref 12, app A), should be changed to read:

8-50 Level Accleration/Obstacle Clearance Takeoff

a. Prior to attempting a level acceleration/obstacle clearance takeoff, thorough performance planning must be accomplished to insure that adequate distance is available to clear the obstacle in question. When using the technique described below, the minimum distance to clear an obstacle should be determined from the takeoff performance chart in chapter 7 using a climbout airspeed of 35 KTAS.

b. Align helicopter with the desired takeoff heading and stabilize at 5-foot main wheel height hover (3 to 4 feet indicated radar altitude). Note aircraft hover pitch attitude on the primary attitude indicator. Apply forward cyclic pressure smoothly while simultaneously increasing collective pitch (if additional power is available within helicopter limits) to begin a level acceleration. The maximum torque available should be applied before accelerating through effective translational lift (ETL) (as noted by abrupt decrease in aircraft vibration and buffet). Additional forward cyclic pressure will be required during the acceleration to overcome the tendency of the aircraft to climb. Maintain the level acceleration until passing through ETL. Once through ETL immediately rotate the helicopter to the hover pitch attitude. Check to insure that maximum available torque is applied. Maintain hover pitch attitude and power setting during climbout. Once the obstacle has been cleared, adjust helicopter attitude and collective pitch, as required, to establish the desired airspeed and rate of climb.

12. The YAH-64 can hover OGE at its maximum alternate gross weight (17,650 pounds) at 5760 feet pressure altitude with standard day conditions and at 1740 feet with a hot day temperature (35°C) (para 9). At more adverse conditions (heavier gross weight, higher altitudes, etc.) takeoff distance can be determined from the takeoff summary data (fig. 6, app E), the power required to hover at a 5-foot wheel height (fig. 2) and the installed engine power available (fig. 1, app D) using the procedure outlined in paragraph 8, appendix D. At the maximum alternate gross weight of 17,650 pounds the YAH-64 in-ground-effect hover ceiling (5 foot wheel height) is 9680 feet at standard day conditions. At these same conditions, a takeoff distance of 952 feet is required to clear a 50-foot obstacle using the recommended climbout airspeed of 35 KTAS. If usable airspeed information was provided to the pilot, so that an airspeed of 20 KTAS could be used, the takeoff distance would be reduced to 727 feet for these conditions.

Climb Performance

Vertical:

13. Vertical climb performance tests were not conducted during this part of the A&FC test program. Vertical climb data were obtained during Part 2 of this A&FC and are presented in reference 7, appendix A. Additional hover data were obtained at various wheel heights and refinements to the OGE hover curve were made. A summary plot of vertical climb performance is presented in figure 12, appendix E. This plot incorporates the increased power available of the YT700-GE-701 engine and reflects the increase in the primary mission gross weight from 14,660 pounds to 14,694 pounds.

14. The vertical rate of climb was calculated at 4000 feet pressure altitude, 35° C, primary mission gross weight of 14,694 pounds, and 95 percent of IRP to be 837 feet/minute. Using 100 percent of IRP under the same atmospheric conditions and gross weight, the vertical rate of climb was calculated to be 1234 feet/minute. The vertical rate of climb exceeded the requirements of paragraph 3.2.1.1.1.1.A. of the system specification (ref 11, app A).

Forward Flight:

15. Continuous climbs were conducted from approximately 1000 to 12,500 feet pressure altitude to determine the power correction factor (K_P) and the weight correction factor (K_W). The climb tests were conducted at the conditions of table 1. An airspeed

schedule for the best rate of climb was determined from level flight performance data and used for all the climb tests. The power schedules used were percentages of IRP based on the test day power available. Test results are presented in figures 13 and 14, appendix E. The K_P was determined to be 0.77. The K_W was found to be a function of thrust coefficient.

Level Flight Performance

16. Level flight performance tests were conducted at the conditions of table 1 to determine power required and fuel flow for airspeeds, altitudes and gross weights throughout the operational envelope of the YAH-64 helicopter. The aircraft was flown at zero sideslip and a main rotor speed of 289 RPM (100 percent). All level flight performance tests were conducted with a dummy 30mm chain gun in the 11 degree elevation, and 0 degree azimuth position without the ammunition chute installed. The dummy Target Acquisition and Designation System (TADS) and Pilot Night Vision System (PNVS) sights were in the stowed position. The HELLFIRE missile launchers were set at the zero degree elevation position. The environmental control unit (ENCU) was operated at a level for pilot and copilot comfort and the stabilator was in the automatic mode. The aircraft in the 8-HELLFIRE configuration at a forward longitudinal center of gravity (cg) was used as a baseline to determine the effects on performance of longitudinal cg changes, and different configurations which included 16-HELLFIRE, clean, and 2-ferry tanks (181 gallon) The effects on power required by high values of referred rotor speed were not investigated because of the permissable power on rotor speed range (98% to 100%) and the prevailing hot (20°C warmer than standard day) atmospheric conditions. Test results are presented nondimensionally in figures 15 through 17, appendix E, for the 8-HELLFIRE configuration and in figures 18 and 19 for the 16-HELLFIRE configuration. Aircraft range and endurance summaries are presented in figures 20 and 21. Dimensional test results are presented in figures 22 through 39. The data presented in figures 25, 26, 27, 29 and 31 were obtained from Part 1 of the A&FC test program and are presented again in this report with the refined fairing that resulted from completion of the data. The total electrical power required by the test aircraft was 1.2 kilowatts which equates to 1.6 horsepower. A list of external items, either installed on the test aircraft and not included in the system specification (ref 11, app A), or defined in reference 11 and not on the test aircraft during this evaluation is presented in table 1, appendix B. The effect of these items was estimated by HHI to reduce the drag of the aircraft by 0.59 ft^2 of equivalent flat plate area. No corrections for electrical load, variable power consumed by the ENCU or external configuration differences between the test and production aircraft were applied to the data in this report.

17. The maximum cruise airspeed using maximum continuous power with the aircraft in the 8-HELLFIRE configuration at the primary mission gross weight of 14,694 pounds and at 4000 feet pressure altitude with an ambient temperature of 35° C and 289 main rotor RPM (100 percent) was determined to be 145 KTAS. A dimensional plot at these conditions is presented in figure 40, appendix E. The aircraft (in the test configuration) met the requirements of paragraph 3.2.1.1.1.1.B of the system specification (ref 11, app A).

18. Long range cruise speeds (based on 99 percent of maximum specific range) and maximum endurance airspeeds were determined for standard day sea level and 4000 feet, 35°C conditions, from figures 15 through 17, appendix E and figures 6 and 7, appendix D. These results are presented in figures 20 and 21, appendix E. The recommended cruise speed for sea level standard day conditions is 125 KTAS for the primary mission gross weight and configuration and 126 KTAS at 4000 feet and 35°C hot day conditions. Maximum endurance speeds for these conditions are 76 and 77 KTAS, respectively.

19. The effects of various sideslip angles on the power required for level flight is presented in figure 22, appendix E. The inherent sideslip of the test aircraft was determined from three flights at different thrust coefficients (C_T) and several airspeeds in the 8-HELLFIRE configuration. These data are presented in figure 23. The inherent sideslip was found to vary with airspeed and C_T. Below a C_T of approximately 0.0090, the inherent sideslip was 2 to 3 degrees right regardless of airspeed. Above a C_T of 0.0090, the inherent sideslip increased as C_T increased. At a C_T of 0.0109 and airspeeds above 100 KTAS, inherent sideslip was approximately 4 degrees right. At a C_T of 0.0109 as airspeed decreased below 100 KTAS the inherent sideslip increased. Varying the sideslip angle from zero to approximately 5 degrees left or 8 degrees right resulted in an increase of drag equal to 0.9 ft² equivalent flat plate area. Using 150 KTAS and sea level standard day conditions, this increases power required for level flight less than 1 percent of dual engine power available (approximately 30 shaft horsepower). Over this range of C_T and airspeeds where aircraft drag is significant the difference in power required for level flight between the zero sideslip trim condition and coordinated ball-centered flight is insignificant.

20. Tests were conducted to determine the change in equivalent flat plate area (Δf_e) with changes from the primary mission configuration and in the longitudinal cg location. The clean, 16-HELLFIRE, and 2-ferry tank (181 gallon) configurations were evaluated. A ferry mission would consist of four ferry tanks

(two on each side), however only two external fuel tanks were available for this test. The Δf_e from the 8-HELLFIRE configuration was not constant for the aircraft configured with 16-HELLFIRE missiles especially at high C_T. Figures 18 and 19, appendix E, present the power required in level flight for the 16-HELLFIRE configuration. Equivalent flat plate area changes between the test configurations and primary mission configuration are presented in table 3. The Δfe for a longitudinal center of gravity change from FS 202.0 to 206.0 (4 in.) in the clean configuration, was -3.1 ft^2.

Table 3. Drag Assessments for Various Configurations

Configuration	Equivalent Flat Plate Area Change (Δfe) from 8-HELLFIRE Configuration (ft^2)
Clean	-8.4
2-Ferry Tank (181 gallon)	-5.4

Autorotational Descent Performance

21. Autorotational descent performance tests were conducted at the conditions of table 1 to determine the effects of gross weight, altitude, airspeed and rotor speed on the rate of descent. Two gross weights and altitudes were evaluated with the aircraft in the 8-HELLFIRE configuration. Coordinated ball-centered flight was used as the trim criteria. The tests were conducted by retarding the power levers to the idle position and then stabilizing the aircraft on an airspeed and rotor speed in autorotation. Airspeed was varied to determine the optimum airspeed for minimum rate of descent ($V_{min\ R/D}$) at the normal operational rotor speed of 289 RPM (100 percent). At the approximate $V_{min\ R/D}$, rotor speed was varied to determine the effect on rate of descent. Test results are presented in figures 41 through 44, appendix E.

22. At a gross weight of approximately 15,300 pounds the $V_{min\ R/D}$ was determined to be 72 knots calibrated airspeed (KCAS) with a rate of descent of 2320 feet/minute (fig. 41, app E). Increasing gross weight approximately 1300 pounds decreased the $V_{min\ R/D}$ to 64 KCAS with an increase in rate of descent to 2430 feet/minute

(fig. 42). The airspeed for maximum glide distance (106 KCAS) could be determined only for the light weight data at 4960 feet density altitude because of the restricted airspeed envelope for the aircraft. Since the airspeed for maximum glide distance at the other conditions tested appeared to be higher than the never exceed airspeed (V_{NE}), 106 KCAS or V_{NE} whichever occurs first should be used for the maximum glide distance airspeed.

23. Rotor speed was varied throughout the permissible operating range of 272 to 301 RPM (94 to 104 percent) to determine the optimum rotor speed for minimum rate of descent (figs. 43 and 44, app E). The rotor speed for minimum rate of descent was determined to be 272 RPM (94 percent) for the conditions tested.

HANDLING QUALITIES

General

24. Stability and control tests were conducted to evaluate the YAH-64 handling qualities with the YT700-GE-701 engine installation and the modified Digital Automatic Stabilization Equipment (DASE) software program. Handling qualities data were obtained at high altitude and in various configurations to supplement data from Parts 1 and 2 of this A&FC. Additional testing included an evaluation of instrument flight capability in light to moderate turbulence and a slope landing evaluation. System specification requirements were also evaluated, where applicable. Both quantitative data and qualitative pilot comments were recorded during these tests. Tests were conducted at the conditions specified in table 2.

Control Positions in Trimmed Forward Flight

25. Control positions in trimmed forward flight were evaluated at the conditions presented in table 2. Data are presented in figures 45 through 51, appendix E. The variation of longitudinal control position with airspeed was conventional except for level flight below 60 KCAS and autorotative flight above 80 KCAS where the position gradient was nearly neutral. In the 8-HELLFIRE asymmetric configuration (fig. 49, app E, and photos 17 and 18, app B), approximately 1 in. of additional right cyclic was required in comparison to symmetrically loaded configurations, however adequate lateral control margin was available and the control positions were not uncomfortable. In climbs and descents at 103 KCAS (fig. 51), lateral and longitudinal trim changes between 0 percent and 100 percent of test day IRP were minimal (0.2 in. and 0.3 in., respectively) and desirable for an attack

helicopter. In the 2-ferry tank configuration (fig. 50), control positions were essentially unchanged from the 8-HELLFIRE configuration at the same longitudinal cg location (fig. 45). For all conditions tested, adequate control margins were available. The control positions in trimmed forward flight are satisfactory. The longitudinal control position variation with airspeed met the requirements of paragraphs 10.3.3.1.2, 10.3.4.1.1, and 10.3.5.2.4 of the system specification (ref 11, app A).

Static Longitudinal Stability

26. The static longitudinal stability characteristics were evaluated at the conditions presented in table 2. The aircraft was trimmed in level, ball-centered flight. The collective was held fixed while airspeed was varied ±20 KIAS in 5-knot increments about trim with the cyclic only. All cyclic inputs were made against the trim feel system to avoid stability augmentation system (SAS) actuator recentering. Data are presented in figures 52 through 54, appendix E. For the low-speed trim point, approximately 60 KCAS, in all configurations, the longitudinal control position gradient was essentially neutral varying less than 0.3 in. over the entire speed range of approximately 40 knots. At the trim point of 95 KCAS or higher, the static longitudinal stability was weakly positive as indicated by the shallow control position gradient. Although the neutral to weak longitudinal static stability contributed to poor trimmability and increased pilot workload in simulated instrument meteorological conditions (IMC), as mentioned in paragraph 46 and A&FC Part 2, (ref 7, app A), it is considered desirable and satisfactory for an attack helicopter operating in visual meteorological conditions (VMC). The static longitudinal stability failed to meet the requirements of paragraph 10.3.4.1 of the system specification (ref 11, app A), in that the variation of longitudinal control position with airspeed was neutral for a level flight trim airspeed of 60 KCAS, but is considered acceptable. The pitch attitude variation with airspeed met the requirements of paragraph 10.3.4.1.2 of reference 11.

Static Lateral-Directional Stability

27. The static lateral-directional stability characteristics were evaluated at the conditions presented in table 2. The aircraft was trimmed in level flight at zero sideslip. The collective control was held constant and sideslip angle was varied in 5-degree increments (left and right) while maintaining constant airspeed and steady heading. Data are presented in figures 55 through 58, appendix E. At 60 KCAS, for all configurations tested, the aircraft exhibited positive directional stability (as indicated

by increased left directional control with increased right sideslip) except at left sideslip angles greater than 20 degrees where directional stability was nearly neutral. Positive dihedral effect (as indicated by increased right lateral control with increased right sideslip) was also exhibited at this trim airspeed. Sideforce characteristics were weak but not significant due to the large sideslip envelope available at low forward airspeed. At a trim airspeed of 95 KCAS and above, for all configurations tested, directional stability and dihedral effect were positive and sideforce cues were significantly increased. For all test conditions no control limits were approached and the static lateral-directional stability characteristics were satisfactory. The static lateral-directional stability characteristics met the requirements of paragraphs 10.3.5.1.5, 10.3.5.1.6 and 10.3.5.1.7 of the system specification (ref 11, app A).

Maneuvering Stability

28. The maneuvering stability characteristics were evaluated at the conditions presented in table 2. The aircraft was trimmed in level, ball-centered flight. The collective was held fixed while cyclic and directional controls were used to perform steady left and right turns at constant airspeed. Data are presented in figures 59 through 61, appendix E. In all configurations tested the longitudinal stick position gradient with load factor was positive. For the 8-HELLFIRE, aft cg configuration at 15,000 feet density altitude, the longitudinal control position gradient with load factor was weaker than it was for the other configurations at lower altitudes, but was approximately 1 in. per g. The maneuvering stability characteristics for all conditions tested are satisfactory. The maneuvering stability characteristics met the requirements of paragraph 10.3.6 of the system specification (ref 11, app A) with the exception of subparagraph 10.3.6.2 which was not evaluated.

Dynamic Stability

29. The longitudinal long term dynamic stability characteristics were evaluated at the conditions presented in table 2. Aircraft motion was induced by displacing longitudinal cyclic from trim and decreasing or increasing airspeed by 10 KIAS then returning the cyclic slowly to trim. All controls were then held fixed until recovery was initiated. Tests were conducted both DASE ON and OFF. Typical time history data are presented in figures 62 and 63, appendix E. With the DASE ON, the long term response was dynamically unstable with a period of between 75 and 90 seconds depending on external configuration. The amplitude of the oscillation increased slowly requiring at least four or more cycles to reach

an airspeed or pitch attitude limit which required the pilot to initiate a recovery. With the DASE OFF, the long term oscillation amplitude increased rapidly requiring recovery in less than two cycles for all configurations tested. The response frequency was faster DASE OFF than DASE ON and increased as oscillation amplitude increased. Recovery back to controlled level flight was accomplished with little difficulty but load factors of greater than 2 g and pitch rates in excess of 25 degrees per second (deg/sec) were experienced. The pilot must closely monitor airspeed and pitch attitude to insure that large pitch rates and excessive pitch attitudes do not occur. The DASE ON long term response characteristics, although unstable, are satisfactory. The DASE OFF long term characteristics are satisfactory for a degraded mode. The long term response characteristics met the requirements of paragraph 10.3.4.2 of the system specification (ref 11, app A).

30. The short term longitudinal and lateral-directional dynamic stability were evaluated at the conditions presented in table 2. Aircraft motion was induced by fore and aft 1 inch cyclic pulses and directional pedal doublets. All inputs were made at an input frequency of approximately one cycle per second. Following the input, all controls were held fixed until the motion subsided or recovery was initiated. All tests were performed with attitude hold OFF and DASE ON and OFF. Typical time history data are presented in figures 64 through 69, appendix E. With the DASE ON, the short term response and lateral-directional oscillations were essentially deadbeat for all conditions tested. With the DASE OFF, the aircraft short term response and lateral-directional oscillations were dynamically stable. The short period motion damped out in three to four cycles, but excited the divergent long period (para 29) causing the aircraft to diverge in pitch. This divergence was of sufficiently long period that it could be easily controlled by the pilot and posed no aircraft control problems during VMC flight. The short term longitudinal and lateral-directional dynamic stability characteristics are satisfactory. The short term longitudinal and lateral-directional dynamic stability met the requirements of paragraphs 10.3.4.2 and 10.3.5.3 of the system specification (ref 11, app A).

Controllability

31. Longitudinal and lateral controllability was evaluated in the 8-HELLFIRE, aft cg configuration, at high density altitude (15,000 ft). Control step inputs were made, using a mechanical control fixture, in 0.25 in. increments up to a maxium of 1.5 in. The inputs were held until a maximum rate was achieved or until recovery was necessary. All tests were conducted with attitude hold OFF and with DASE both ON and OFF.

32. Longitudinal controllability characteristics at 80 KCAS are summarized in figure 70, appendix E. The maximum longitudinal control response was 17 deg/sec/in. for forward and 18 deg/sec/in. for aft inputs DASE ON and OFF. Control sensitivity was greater DASE ON (27 $deg/sec^2/in.$ forward and 39 $deg/sec^2/in.$ aft) than DASE OFF (14 $deg/sec^2/in.$ forward and 28 $deg/sec^2/in.$ aft) and the average response time (time to reach 63 percent of maximum rate) was 0.5 sec with DASE ON and 1.2 sec with DASE OFF. Aircraft response was faster than noted during A&FC, Part 2 (ref 7, app A) for the same configuration at a density altitude of 5000 ft but no handling qualities problems were noted. Typical time history data are presented in figures 71 and 72. The longitudinal controllability characteristics are satisfactory. The longitudinal controllability met the requirements of paragraphs 10.3.4.3 and 10.3.4.4.1 of the system specification (ref 11, app A). The longitudinal controllability failed to meet the requirements of paragraph 10.3.4.4.2 of reference 11 in that the average response time to a longitudinal control step input was less than 0.7 sec by 0.2 sec, but is acceptable.

33. The lateral controllability characteristics at 80 KCAS were essentially the same DASE ON and OFF. These characteristics are summarized in figure 73, appendix E. Control response was 30 deg/sec/in. for left and right inputs while control sensitivity was 66 $deg/sec^2/in.$ for left and 70 $deg/sec^2/in.$ for right inputs. Average response time was 0.4 sec. Roll response was slightly more rapid than that noted during A&FC, Part 2 (ref 7, app A) for the same configuration at a density altitude of 5000 ft. Roll response was more rapid than pitch response and gave excellent maneuver capability which is desirable for an attack helicopter. Typical time history data are presented in figures 74 and 75. The lateral controllability is satisfactory. The lateral controllability met the requirements of paragraphs 10.3.5.2.2, 10.3.5.2.3, and 10.3.5.2.5 of the system specification (ref 11, app A). The lateral controllability with DASE ON failed to meet the requirements of paragraph 10.3.5.2.1 of reference 11 in that the average response time to a lateral control step input was less than 0.7 sec by 0.3 sec, but is acceptable.

Ground Handling Characteristics

34. Operation of the parking brake was evaluated during this test program. During EDT-4 and earlier tests the lack of a reliable indication of parking brake status had been reported as a shortcoming because the parking brake handle could be fully retracted with the brake still set. During this test it was also noted that the parking brake handle would remain in the extended position even though the brakes were not set. The lack of a reliable

indication of parking brake status remains a shortcoming. The following caution should be placed in chapter 2 of the operator's manual:

CAUTION

Do not rely on the position of the parking brake handle as an indication of parking brake status. Brakes must be reset or released, as appropriate, to determine correct status of parking brake.

Slope Landing Characteristics

35. The slope landing capabilities of the YAH-64 were evaluated at the conditions presented in table 2. The landings and takeoffs were made on a surveyed slope constructed at Yuma Proving Ground, Arizona. The slope area was divided into four segments of, nominally, 8 degrees, 10 degrees, 12 degrees, and 15 degrees to facilitate the evaluation. The surface was hard packed sandy clay with 23 percent one-half inch gravel with some rocks 3 inches in diameter. The soil was classified as "SC" under the Unified Soil Classification System (ref 15, app A). The actual slope angle of the main landing gear was measured with an inclinometer after the aircraft departed the slope area. Prior to the test, the aircraft fluid levels and landing gear struts were serviced in accordance with HHI maintenance procedures. Each main transmission sump was overserviced by one gallon in accordance with AVRADCOM Memorandum, DRDAV-DP, dated 4 August 1982. All tests were conducted in winds of 5 knots or less. The technique employed was a vertical landing and takeoff which was repeated for each slope orientation tested. The parking brake was set and the tail wheel was locked during all tests. All landings were made vertically from a hover. When ground contact was made, the collective was slowly lowered while cyclic was held into the hill to keep the aircraft from sliding downslope. When sufficient weight was on all three landing gear, the cyclic was slowly centered and the collective lowered to full down. Landing was complete when the cyclic was fully centered, the collective full down, and directional pedals set at the position corresponding to zero tail wheel side load on a level surface. Takeoffs were made by simultaneously increasing collective and displacing lateral cyclic toward the hill. As the aircraft began to lift off, cyclic was adjusted, as required, to make a vertical takeoff to a hover. Aircraft attitudes were measured by an on-board instrumentation system. Photographs were taken of all landings and takeoffs.

36. Lateral slope landings up to 10-1/2 degrees left wheel upslope and 9-1/2 degrees right wheel upslope were made. Data are presented in figures 76 and 77, appendix E. Adequate control margins remained in all directions and the aircraft did not slide down slope for any angle tested. Centering of the cyclic was performed slowly due to the high fuselage roll angles resulting from differential main landing gear strut compression and the narrow main landing gear wheel base. As the cyclic was centered, strut compression was jerky rather than a smooth continuous movement. This motion combined with large fuselage roll attitudes for slopes greater than 9 degrees gave the sensation of impending roll-over to the crew. Fuselage roll attitude was 17 degrees initially for the 10-1/2 degree left gear upslope landing. Once the cyclic had been centered and the collective placed full down the aircraft settled (due to tire and strut compression) and the fuselage roll attiude reached 19 degrees. These large fuselage roll attitudes produced uncomfortable side loads on the crew and also resulted in high side loads on the main landing gear tires (photos 3 through 5). It appeared that separation of the tire from the rim could occur when landing on slopes of greater than 9 degrees with fuselage roll angles in excess of 12 degrees. Figure A shows the fuselage roll attitude associated with slope angles ranging from zero to 10-1/2 degrees. Lateral slope landing characteristics, on lateral slopes of up to 9 degrees left and right, were satisfactory. The zero wind lateral slope landing envelope should be limited to 9 degrees maximum slope angle and 12 degree fuselage roll attitude. The lateral slope landing characteristics failed to meet the 15 degree slope landing requirements of paragraph 10.3.10.6 of the system specification (ref 11, app A) due to large fuselage roll attitudes and the possibility of tire/rim separation during operation on a lateral slope of 10-1/2 degrees. The requirement for a slope landing capability on greater than 9 degree slopes should be re-evaluated by the user. The following caution should be placed in chapter 8 of the operator's manual.

CAUTION

During slope landing operations, use the primary attitude indicator to insure that fuselage roll attitude does not exceed 12 degrees when centering the cyclic. Should fuselage roll attitude reach 12 degrees before the cyclic has been centered, reposition the aircraft to a shallower slope.

37. Longitudinal slope landings up to 12 degrees nose up and 10 degrees nose down were made. Time history data are presented

Photo 3, 10 1/2 Degree Slope Landing, Left Gear Upslope, Front View

Photo 4. 10 1/2 Degree Slope Landing, Left Gear Upslope,
Right Rear Quartering View

FIGURE A
AIRCRAFT ROLL ATTITUDE VERSUS GROUND SLOPE ANGLE DURING SLOPE LANDINGS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)		AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AIRCRAFT CONFIGURATION
	LONG (FS)	LAT (BL)				
14,480	206.8(MID)	0.8(LT)	2780	29.0	290	CLEAN

NOTES: 1. AIRCRAFT STATIONARY ON THE GROUND, ROTORS TURNING AND CYCLIC CONTROL CENTERED.
2. ▣ - MAXIMUM ROLL ATTITUDE RECORDED IMMEDIATELY AFTER LANDING
3. ⊙ - MAXIMUM ROLL ATTITUDE RECORDED WITH THE AIRCRAFT ON THE GROUND AFTER SETTLING OCCURRED

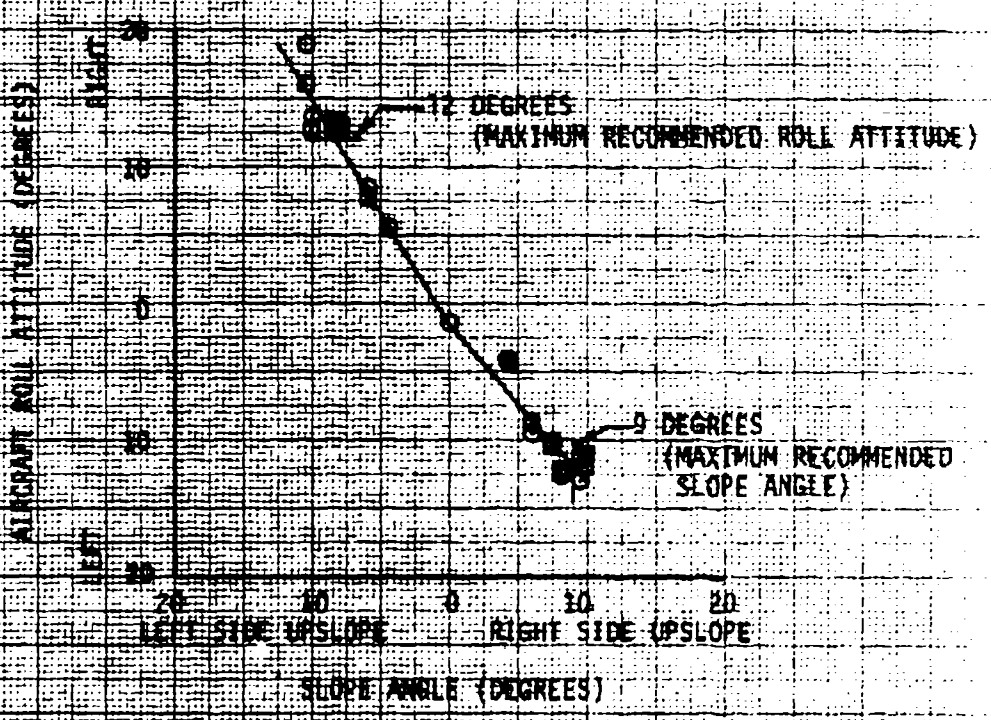


Photo 5. 10 1/2 Degree Slope Landing, Left Gear Upslope,
Right Front Quartering View

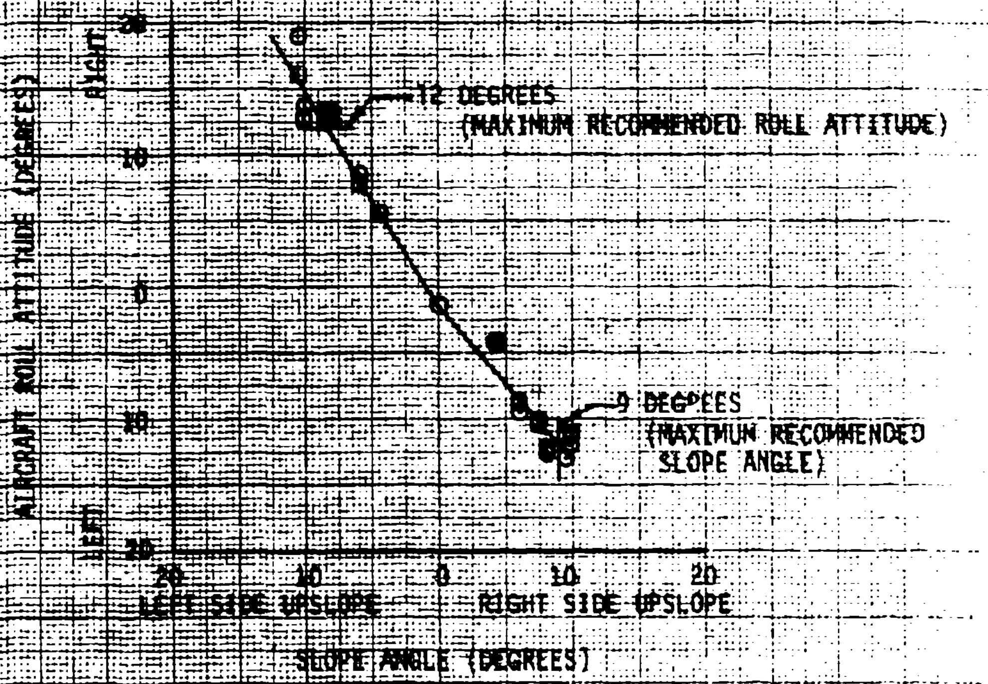
FIGURE A

AIRCRAFT ROLL ATTITUDE VERSUS GROUND SLOPE ANGLE DURING SLOPE LANDINGS

YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AIRCRAFT CONFIGURATION
14,480	204.8(MID)	0.8(LT)	2780	29.0	290	CLEAN

NOTES: 1. AIRCRAFT STATIONARY ON THE GROUND, ROTORS TURNING AND CYCLIC CONTROL CENTERED.
2. @ - MAXIMUM ROLL ATTITUDE RECORDED IMMEDIATELY AFTER LANDING
3. @ - MAXIMUM ROLL ATTITUDE RECORDED WITH THE AIRCRAFT ON THE GROUND AFTER SETTLING OCCURRED



in figures 78 and 79, appendix E. For all landings, the aircraft brakes held and no downhill tire sliding occurred. Control margins were adequate for landings and takeoffs from slopes up to 10 degrees nose up and down. During the 12 degrees nose up landing, droop stop contact resulted in impact damage to the striker plates and rollers and rotor blade canopy clearance was less than nine inches as indicated by the main rotor blades breaking the nine inch canopy clearance stick (photo 6). Throughout the landing there was no aural or control feedback indication to the pilot that droop stop contact was made. The slope landing characteristics, on longitudinal slopes of up to 10 degrees nose up and nose down, are satisfactory. The longitudinal slope landing envelope should be limited to 10 degrees maximum slope angle. The slope landing characteristics for longitudinal slopes failed to meet the 12 degree slope landing requirements of paragraph 10.3.10.6 of the system specification (ref 11, app A) in that the main rotor droop stops were damaged when landing on a 12 degree longitudinal slope. The requirement for a slope landing capability on greater than 10 degree longitudinal slopes should be re-evaluated by the user.

38. Tests were conducted to provide data for calculation of wing store clearance with various external store and pylon installations. Minimum wing store clearance was determined using 16mm movies of lateral slope landings to measure the vertical distance from the outboard wing pylon stores pivot point to the ground. Figure B shows the physical measurement taken from the test aircraft between the forward, outboard pylon attaching point and the stores pivot point. Clearances for the upslope pylon were measured during takeoffs and landings. Clearances for the downslope pylon were measured with the aircraft settled on the slope with the collective full down and cyclic centered. The data presented in figure C shows the minimum clearance between the outboard pylon stores pivot point and the ground on the test aircraft. The line showing HELLFIRE missile ground contact was determined using measurements taken from the test aircraft configured with 4-HELLFIRE missiles on each outboard pylon set at zero degrees elevation. This configuration may not be representative of the production aircraft.

Power Management

39. The power management characteristics of the YAH-64, configured with the YT700-GE-701 engines, were evaluated during this test program. Specific tests included evaluations of the contingency power feature of the engines, engine/airframe response characteristics, and engine functional checks. Particular emphasis was placed on operation at power levels above those attainable with

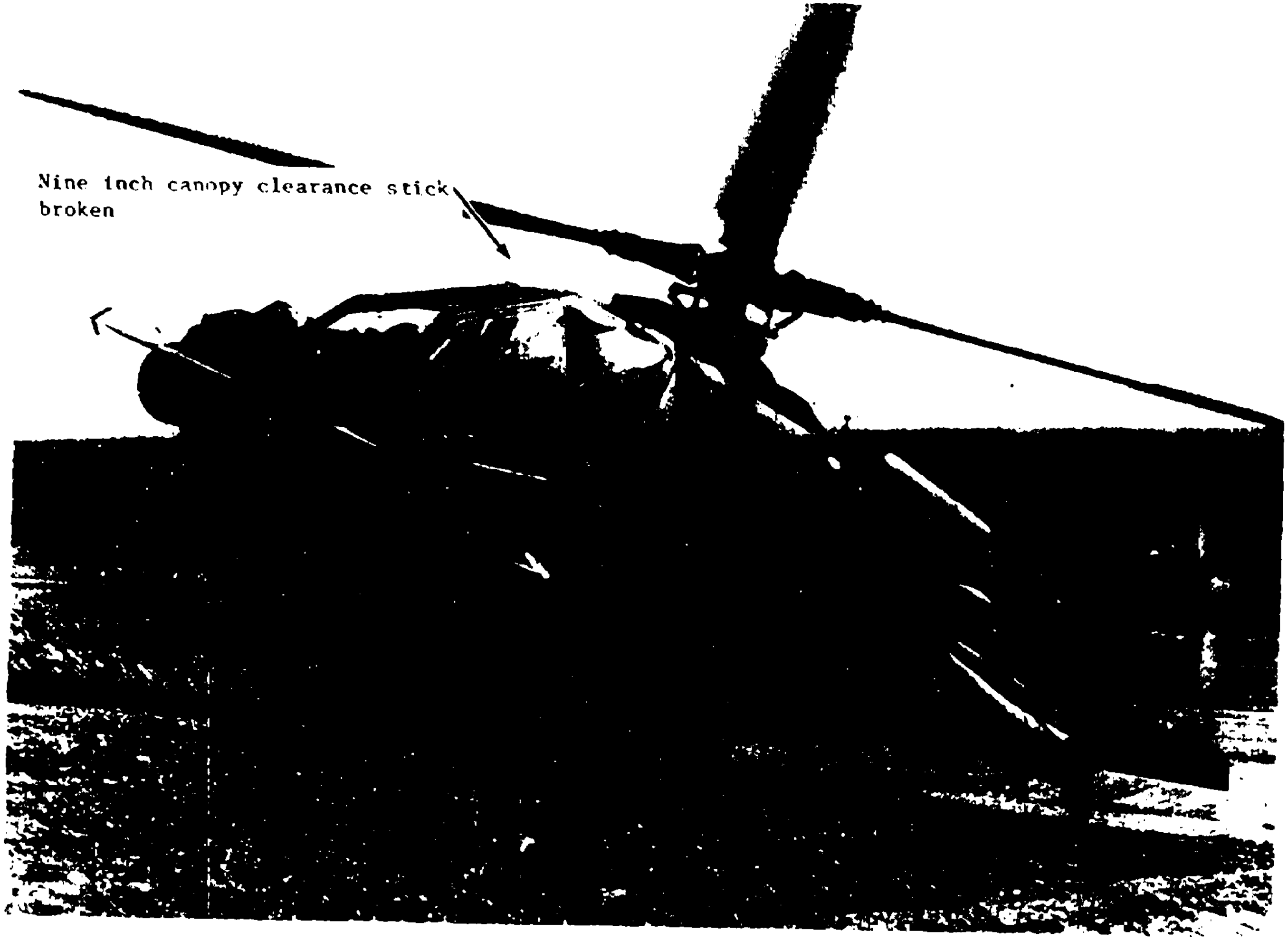


Photo 6. 12 Degree Slope Landing, Nose Upslope, Left Front Quartering View

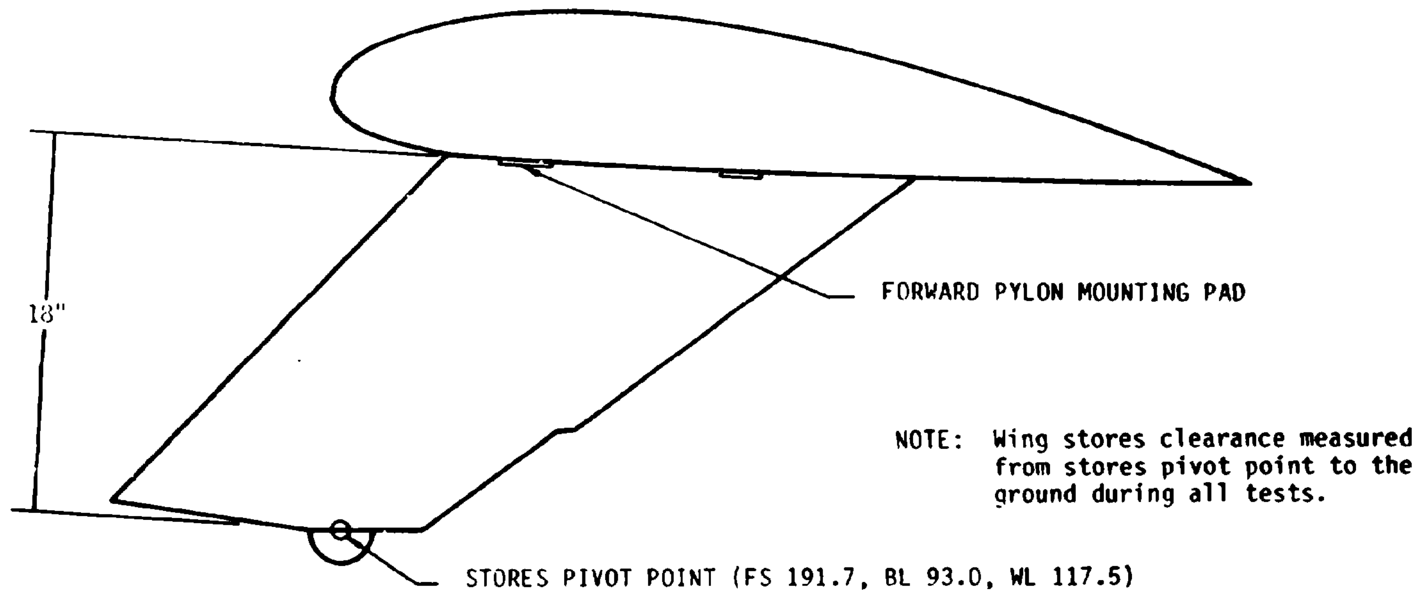
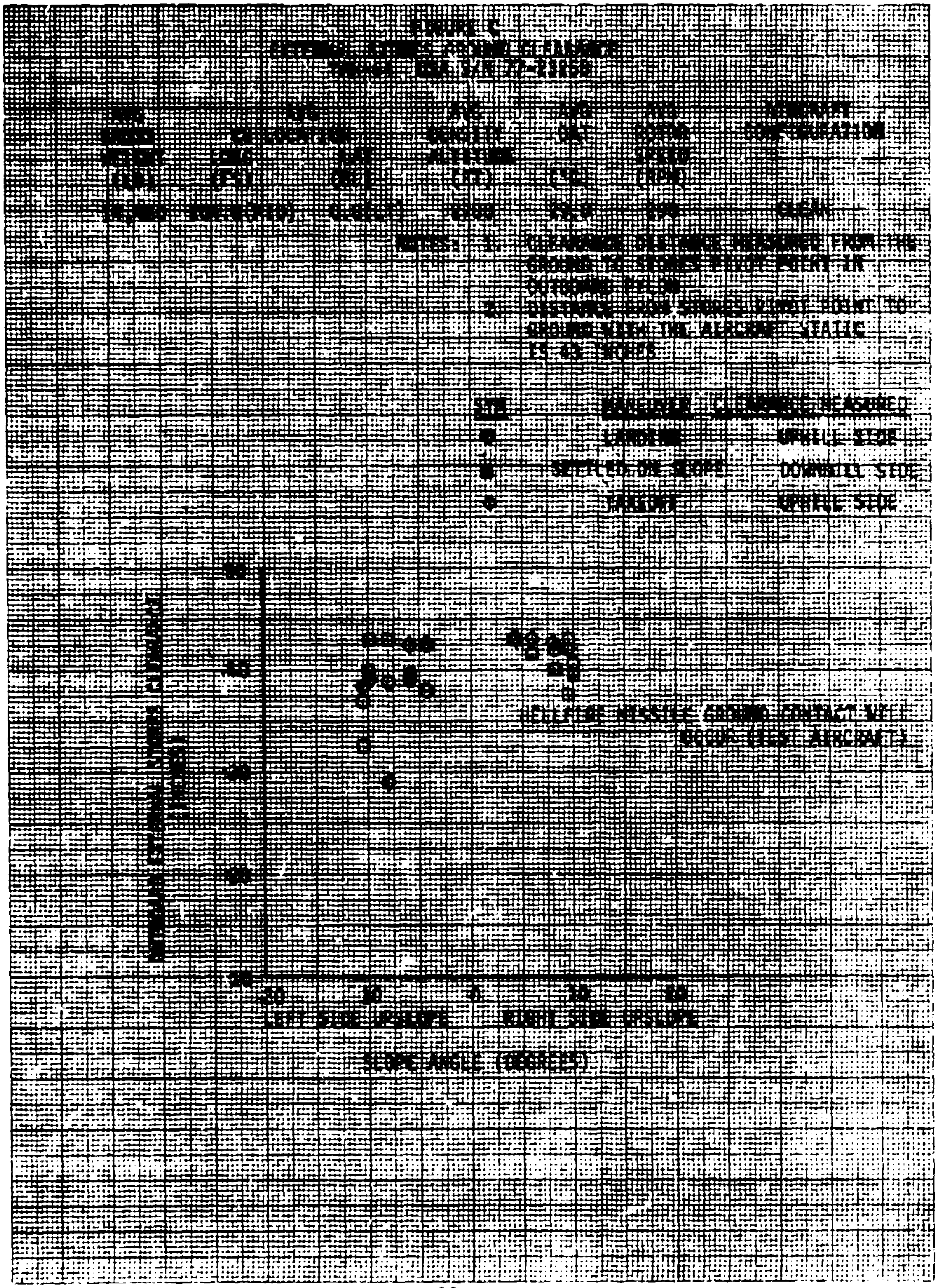


FIGURE B.
OUTBOARD PYLON GEOMETRY


NOTES:
GROUND WITH THE AIRCRAFT STATIC
SYM
MANEUVER
CLEARANCE MEASURED
LANDING
UPHILL SIDE
SETTLED ON SLOPE
DOWNHILL SIDE
TAKEOFF
UPHILL SIDE
HELLFIRE MISSILE GROUND CONTACT WILL
OCCUR (TEST AIRCRAFT)
20
10
0
10
20
LEFT SIDE UPSLOPE
RIGHT SIDE UPSLOPE
SLOPE ANGLE (DEGREES)

the previous YT700-GE-700R engines. No compressor stalls were encountered during high altitude testing, at Bishop, California (elevation 9980 feet), as previously reported for the YT700-GE-700R engines (ref 6, app A). Generally, the YT700-GE-701 engines provided a significant increase in power available, compared to the YT700-GE-700R engines, and proved to be very reliable throughout the test program.

40. The contingency power feature of the YT700-GE-701 engines is automatically activated when one engine fails. It provides increased single engine power available by automatically resetting the turbine gas temperature (TGT) limiting threshold from 867 to 917°C. At conditions where engine power available was transmission limited, this resulted in an increased power available of 22 percent above IRP, on the operating engine. The use of contingency power was limited to 2.5 minutes maximum by reference 10, appendix A. Figure 80, appendix E is a time history of a simulated single engine failure during an OGE hover showing the additional torque available during single engine operation. The transition to contingency power was smooth and consistent and required no pilot action. This feature significantly increased the single engine capability of the aircraft and will enhance survivability in an operational environment. The automatic contingency power feature of the YT700-GE-701 engine is an enhancing characteristic.

41. The engine/airframe response characteristics were evaluated at the conditions shown in table 2. Tests included various rates of collective control application and reduction between autorotation and IRP, collective control reversals, and a qualitative evaluation of mission representative maneuvers. Time history data are presented in figures 81 through 83, appendix E. The most significant response noted was for a collective control increase of 4.7 in. at a rate of 1.5 in. per second. When initiating this collective control input from 10 percent indicated engine torque, no adverse engine/airframe response was noted (fig. 81). When initiating the same control input from a zero torque condition, main rotor speed droop and activation of the low main rotor warning was observed along with subsequent engine/airframe oscillations (fig. 82). The oscillations damped in approximately 12 seconds and within 4 cycles. Main rotor speed (N_R) varied from 94 to 105 percent (maximum transient 104 percent) and yaw rates oscillated between ± 12 deg/sec. This response was undesirable and distracting to the pilot and required that he reduce the severity of the maneuver and direct his attention inside the cockpit while attempting to compensate for the main rotor speed fluctuations and aircraft oscillations (HQRS 5). The response characteristics noted during this test

qualitatively appeared to be degraded from that observed during previous evaluations with the YT700-GE-700R engines installed. The engine/airframe response observed during this test will be most prevalent during typical NOE maneuvers such as a quick stop (fig. 83) and will degrade the maneuvering capability of the aircraft. The undesirable engine/airframe response during power applications from a zero torque condition is a shortcoming which should be corrected prior to operational use of the helicopter.

42. The engine response to variations in main rotor speed was evaluated during a low power, gradual deceleration from 100 KIAS. Collective control position remained fixed throughout the maneuver. Time history data of this maneuver are presented in figure 84, appendix E. The deceleration resulted in an increase in N_R to 103.5 percent followed by a decrease to 97.7 percent. As N_R increased, engine fuel flow began to decrease at approximately 100.5 percent N_R. However, as main rotor speed decreased, a corresponding increase in fuel flow was not observed until N_R had reached 98.6 percent. The engine response to variations in main rotor speed, though not a problem during this test, may have contributed to the undesireable engine/airframe response discussed in paragraph 41 and should be further investigated.

43. Functional checks of the engine controls, monitoring systems and checklist procedures were performed during this test and were evaluated qualitatively. Tests included operational checks of engine start, runup and shutdown procedures, engine overspeed system and electrical control unit (ECU) lockout procedures. Qualitative evaluations of the TGT limiting system and torque matching characteristics were also conducted. All of these systems and procedures were satisfactory to the extent of the evaluation.

Instrument Flight Capability

44. A limited IMC evaluation was performed consisting of two flights in light-to-moderate turbulence. Turbulence reporting criteria are defined in table 1, appendix D. Maneuvers performed included climbs, descents, climbing and descending turns, changing airspeed and retrimming in level flight, simulated instrument approaches and instrument takeoffs. IMC conditions were simulated by the installation of white curtains in the pilot station as shown in photos 7 through 9. All flight instruments were used including the EADI. Instrument approaches were done using Precision Approach Radar at Marine Corps Air Station, Yuma, Arizona and using the doppler to simulate Automatic Direction Finder stations. The aircraft flight instruments are shown in photo 10. The PNVS was not operational during this

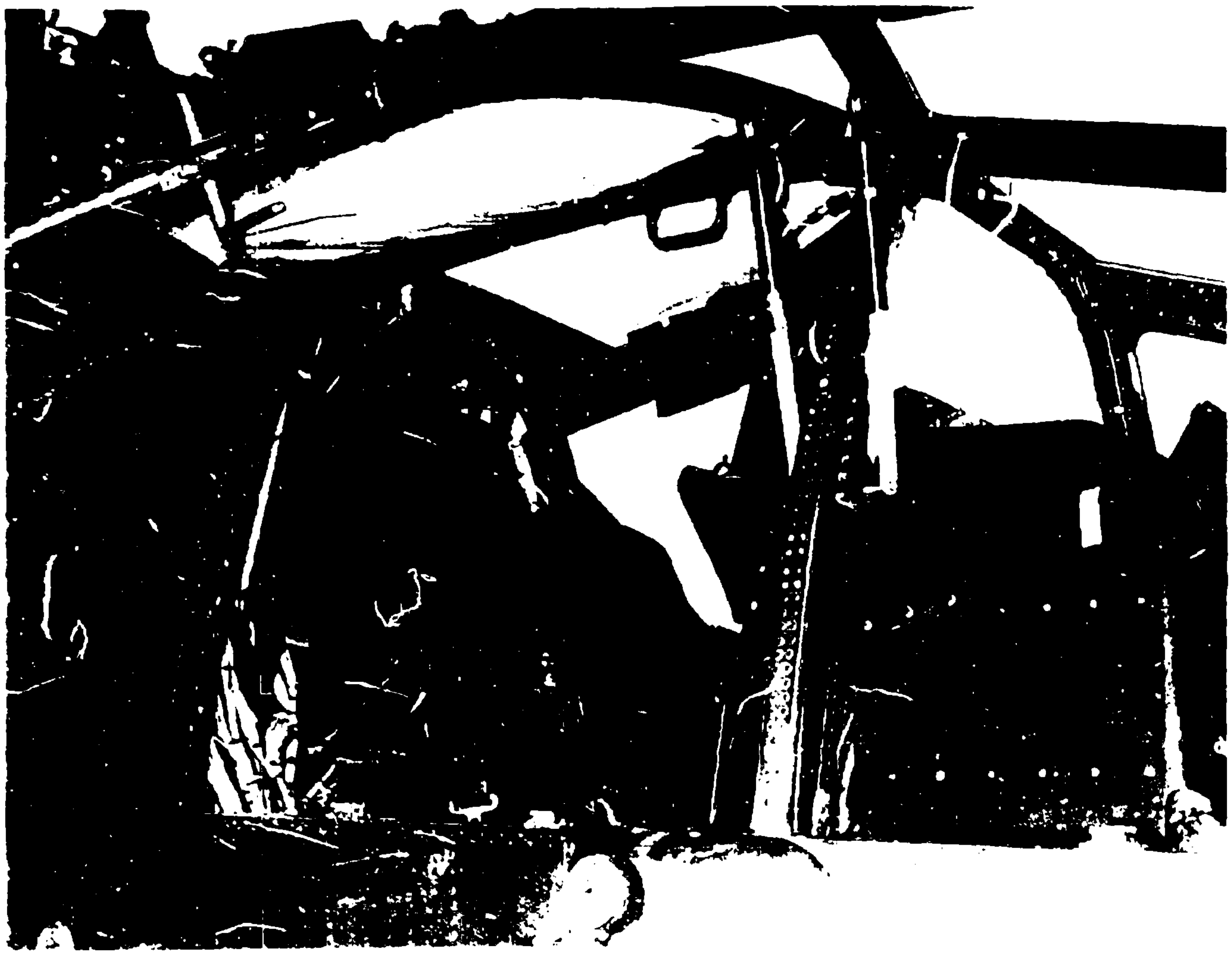


Photo 7. PPC Curtain Installation (Interior) Right Front View

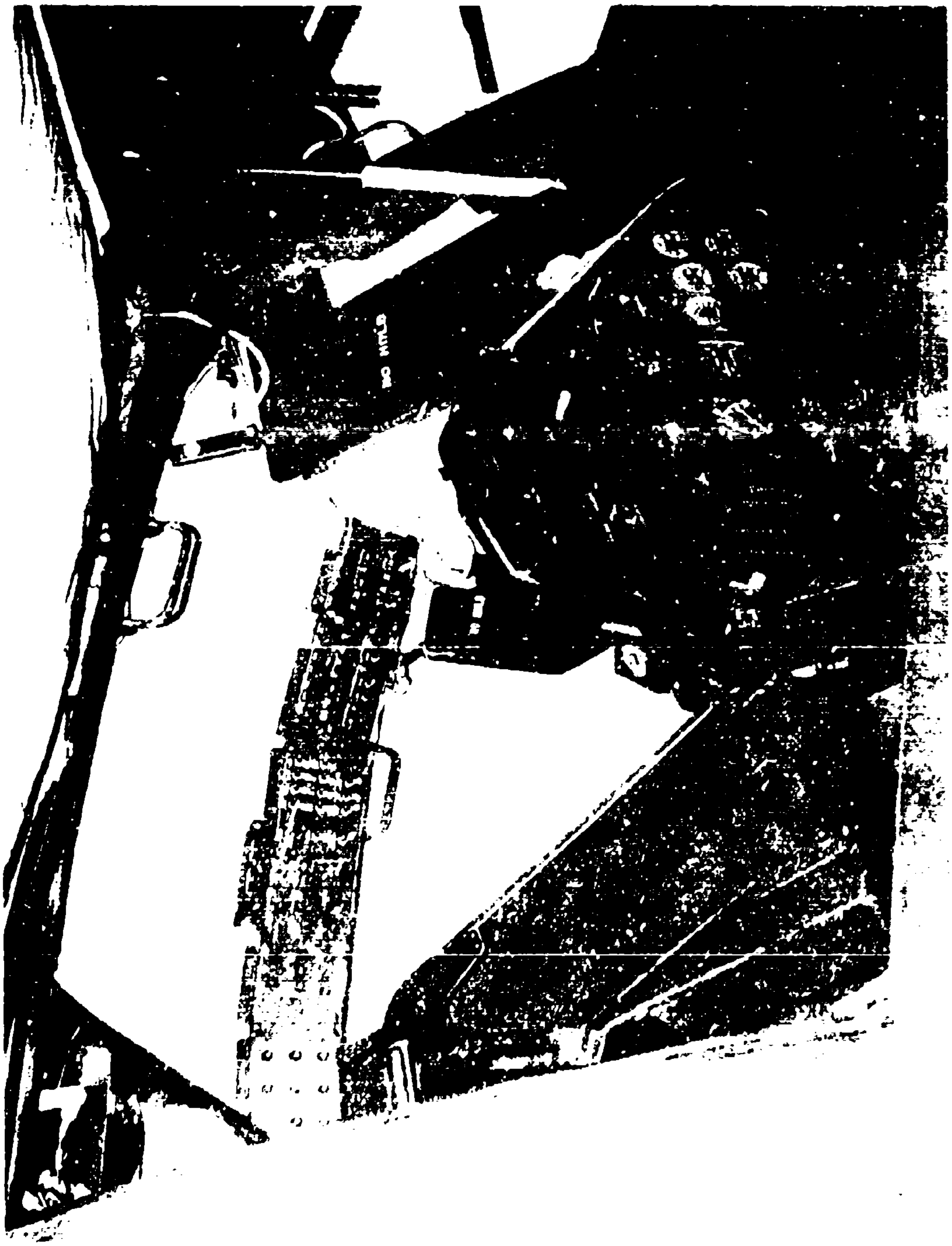


Photo 9. TMC Curtain Installation (Exterior)

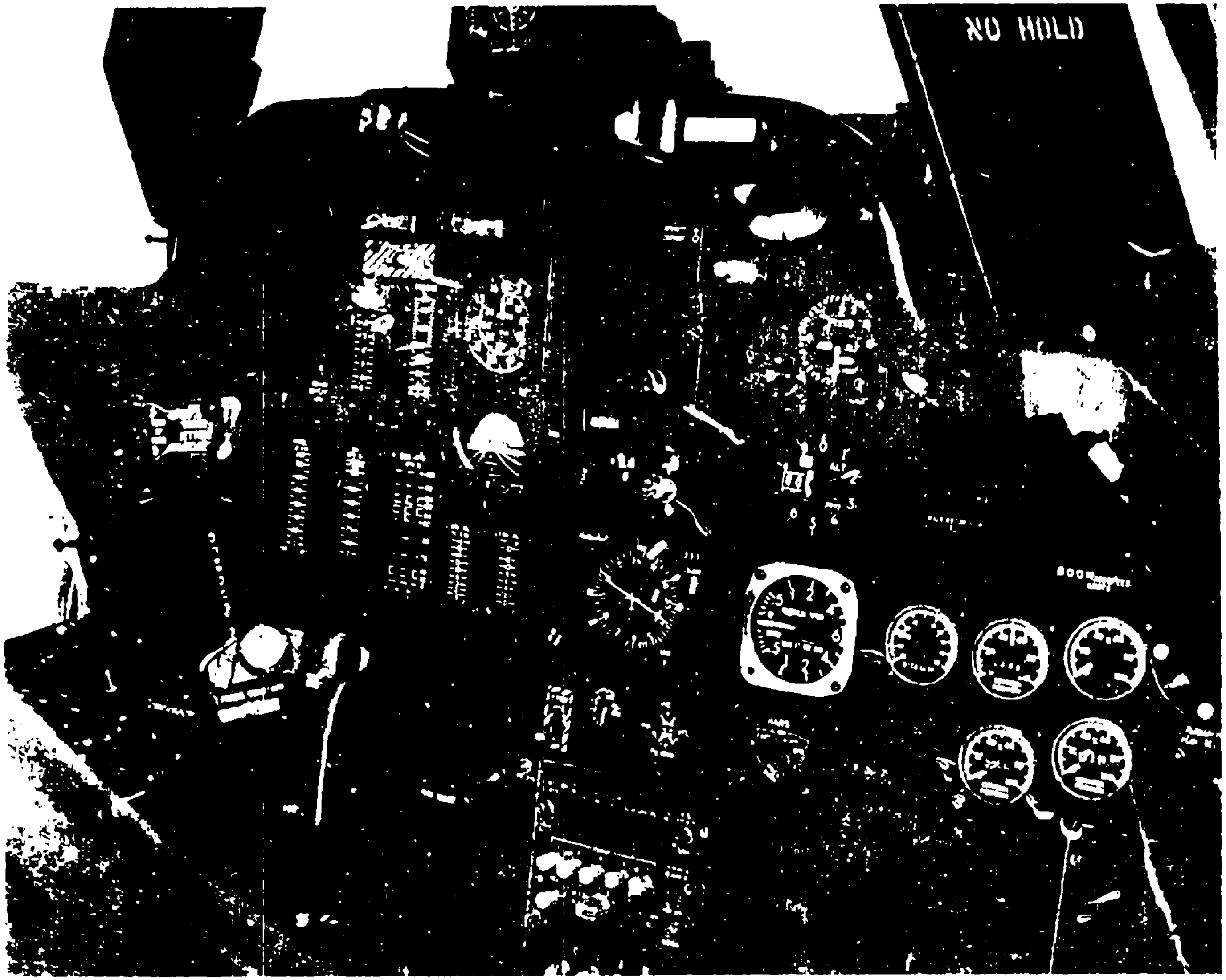


Photo 1[illegible], Pilot Instrument Panel

test. The production airspeed configuration (left wing pitot tube connected to the pilot airspeed indicator) was used for the first flight only. Maneuvers were flown DASE ON with the attitude hold ON and OFF and with DASE OFF (DASE must be ON to allow attitude hold to be engaged).

45. With the attitude hold engaged the aircraft pitch attitude remained within ± 3 degrees of trim in light to moderate turbulence with only minimal pilot input (HQRS 3). Upon initial engagement of attitude hold a pitch oscillation of ± 2 to 3 degrees degrees was experienced which damped out in two to three cycles. When retrimming with attitude hold ON, the recentering of the SAS actuators caused changes in the aircraft attitude, primarily in pitch. Several large longitudinal cyclic inputs were required to set the proper pitch attitude and achieve the desired airspeed (HQRS 4). Time history plots of retrimming with attitude hold ON (figs. 85 and 86, app E) show longitudinal cyclic inputs of up to 1.5 in. required to counter the recentering of the SAS actuators. This contributed to increased pilot workload required to retrim the aircraft in turbulence and appeared, qualitatively, to be degraded from A&FC, Part 2 (ref 7, app A). Because of this problem the pilot tended to fly against the force gradients rather than retrim when small airspeed changes were required. The excessively large longitudinal cyclic inputs required when retrimming, during IMC flight with attitude hold ON, is a shortcoming.

46. With the DASE ON and attitude hold OFF, pilot workload increased significantly, particularly in pitch control, to maintain airspeed within ± 10 KIAS (HQRS 5). The weak longitudinal static stability (para 26) contributed to poor trimmability requiring the pilot to continuously retrim the aircraft. The trim system freeplay (0.3 in.) noted during A&FC, Part 2 (ref 7, app A), also added to the trimming task as the cyclic tended to creep forward requiring the pilot to set the trim aft of the required trim position to compensate. The IMC flight characteristics with DASE ON and attitude hold OFF are satisfactory since this is considered a degraded mode of operation. The trim system failed to meet the requirements of paragraph 10.3.2.5 of the system specification (ref 11, app A) in that the cyclic control would not maintain the zero force position selected by the pilot.

47. With the DASE OFF, pilot workload was extremely high, particularly in pitch where the divergent long term characteristics (para 29) made airspeed and pitch attitude control difficult (HQRS 6). Aircraft control was still possible and several approaches were flown in this condition. Airspeed could not be

maintained within ±10 KIAS in turbulence at all times. This made altitude control difficult, particularly during descent on instrument approaches and large collective changes were required to try to maintain a reasonable approach path (HQRS 5). Although difficult to fly, the aircraft was controllable and a successful instrument approach could be accomplished. The IMC flight characteristics with DASE OFF are satisfactory for a degraded mode of operation. DASE OFF flight under simulated IMC conditions met the requirements of paragraph 10.3.2.7.3 of the system specification (ref 11, app A).

Digitial Automatic Stabilization Equipment Evaluation

48. Tests were conducted to evaluate the production configuration DASE software (ref 17, app A) which was installed in July 1982 prior to the handling qualities portion of the A&FC. The production DASE configuration is described in appendix B. Tests were conducted to assess the gust response in forward flight and hover with the DASE ON, with attitude hold ON and with Hover Augmentation System (HAS) ON. Additional tests were done, as recommended in A&FC, Part 2, (ref 7, app A), to investigate aircraft response to sudden recentering of the SAS actuators (simulated DASE failures) and to determine if the divergent pitch and roll oscillations in HAS mode had been corrected. Test conditions are shown in table 2.

49. The level flight gust response was evaluated in light and occasionally moderate turbulence with the DASE ON and attitude hold ON and OFF. The pilot workload required to maintain airspeed within ±10 KIAS was not noticeably different from A&FC, Part 2 (ref 7, app A) and the gust response appeared unchanged. In turbulence with attitude hold ON, the attitude remained within ±3 degrees from the trim condition with minimal pilot input (HQRS 3). The forward flight gust response, with attitude hold ON and OFF, is satisfactory.

50. The gust response in a hover was evaluated in gusty winds of 8 to 14 knots, with the DASE ON and the HAS ON and OFF. Tests were conducted in head wind, tail wind, and crosswind conditions. With the HAS ON the aircraft held the hover position within 20 feet with only occasional pilot input (HQRS 3). On several occasions, after the aircraft was trimmed at a hover, HAS engaged and cyclic released, the aircraft started to drift and accelerate away from the desired hover location. The drift was easily arrested and after retrimming the HAS held the aircraft over the desired hover location. The drift appeared to occur only upon initial HAS engagement and not after retrimming with the HAS already engaged. HAS inputs were not abrupt as in earlier tests

but were similar to those of a pilot operating under similar conditions. The divergent pitch and roll oscillations experienced with the HAS in A&FC, Part 2 (ref 7, app A) were not present in the new DASE configuration. The gust response in hovering flight, with the DASE ON and the HAS ON and OFF, is satisfactory. The previously reported shortcoming of divergent pitch and roll oscillations while hovering with the HAS engaged has been eliminated.

51. Simulated three axis DASE failures were performed during level flight, with attitude hold ON and OFF, by depressing the DASE release switch on the pilot cyclic grip. Prior to disengagement of the DASE, the aircraft was stabilized in trim for one minute. After disengagement, all controls were held fixed for a minimum of three seconds. Time history data from these tests are presented in figures 87 through 90, appendix E. With attitude hold ON, the SAS actuators were displaced farthest from center giving the most abrupt response when centering occured. For all level flight disengagements, pitch, roll, and yaw rates were 5 deg/sec or less after three seconds and no control problems existed as evidenced by the pilot being able to hold all controls fixed for a minimum of six seconds before recovery. The aircraft response to simulated DASE failures, in forward flight with attitude hold ON and OFF, is now satisfactory. The aircraft response to abrupt disengagement of the automatic stabilization equipment in level flight met the requirements of paragraph 10.3.2.7.1 of the system specification (ref 11, app A).

52. Simulated three axis DASE failures were performed in a hover with HAS ON and OFF using the same technique as described in paragraph 51. In addition, simulated failures were performed, with HAS ON, by attempting to drive the actuators to saturation in one axis with control inputs and then disengaging the DASE. Time history data for these tests are presented in figures 91 through 94, appendix E. During A&FC, Part 2 (ref 7, app A) it was suspected that abrupt recentering of the SAS actuators from a saturated condition, following a DASE failure, could be a significant problem during NOE flight and confined area operations. When driving the actuators to saturation and then disengaging the DASE, aircraft response was abrupt, however, with the pilot in-the-loop, control was maintained and the aircraft was easily re-established in a stabilized hover (HQRS 3). Forward saturation of the longitudinal SAS actuator, which was considered worst case due to the 20% control authority, was attempted but could not be accomplished (fig. 93). Due to the large stick inputs required to come close to forward SAS saturation with HAS ON, it was considered unlikely that this condition could occur under

mission conditions. Aircraft response after DASE disengagement from a one minute stabilized hover resulted in pitch, roll, and yaw rates of less than 3 deg/sec within three seconds (fig. 94). The controls were held fixed for over seven seconds and then only small control movements were required to re-establish a stabilized hover. The aircraft response to simulated DASE failures, in hovering flight with the HAS ON and OFF, is now satisfactory. Aircraft response to abrupt disengagements of the automatic stabilization equipment in a hover met the requirements of paragraph 10.3.2.7.1 of the system specification (ref 11, app A).

53. During A&FC, Part 2 (ref 7, app A) it was determined that due to long SAS and Command Augmentation System (CAS) washout times, rate damping could be reduced during flight in turbulent conditions due to SAS actuator saturation. An instrument takeoff maneuver similar to that flown during A&FC, Part 2 which identified this possibility, was flown in light to moderate turbulence and showed that at all times during the maneuver, rate damping authority was adequate (fig. 95, app E). During this and other mission maneuvers flown in turbulence, the SAS actuators saturated only occasionally and remained capable of providing adequate rate damping in pitch, roll, and yaw throughout each maneuver. The occasional SAS actuator saturation produced no undesirable handling qualities and was not noticeable to the pilot. The previously reported shortcoming of a reduction of rate damping authority caused by excessive SAS and CAS washout times has been eliminated.

Simulated Single Engine Failures

54. Single engine failures were simulated by having the copilot/gunner (CPG) retard one of the power levers to the idle position. Tests were conducted at the conditions shown in table 2. Collective control reduction was made after 2.0 seconds delay time or 1.0 second after the activation of the low main rotor speed warning, for specification compliance. Representative time histories of simulated single engine failures are presented in figures 96 and 97, appendix E. Initial aircraft response was a slight left yaw which was easily corrected by instinctive directional control inputs (HQRS 2). Due to this mild aircraft response, the first indication of an engine failure may often be the engine out/low main rotor speed audio tone. A collective control delay of 1.0 second after activation of the low main rotor speed warning resulted in N_R droop to a minimum of 87.3 percent during a simulated engine failure at maximum horizontal velocity (V_H) (fig. 96). This was 5.7 percent below the minimum power-on transient N_R of 93 percent. No adverse handling qualities were observed at this low rotor speed and, upon collective control

reduction, N_R returned to the normal operating range (98 to 100 percent) within 3 seconds. Immediate reduction of the collective control, following activation of the low main rotor speed warning, resulted in a minimum N_R of 92.3 percent and a total collective control delay time of only 1.2 seconds at V_H (fig. 97). During an actual engine failure at high power settings, rapid N_R decay rates and pilot reaction times in excess of one second may cause N_R to droop below 93 percent. Although aircraft response to simulated single engine failures appears satisfactory to the pilot, single engine operation at contingency power below a transient N_R of 93 percent should be investigated to insure that aircraft structural load limits are not exceeded. The aircraft response to simulated single engine failures failed to meet the requirements of paragraph 10.3.8.1.1 of the system specification (ref 11, app A) in that the available collective control delay time at V_H was less than 2.0 sec by 0.8 sec.

55. The engine out warning system was evaluated during this test to determine if a false indication of a dual engine failure in the event of a single engine failure would occur. This was a previously reported deficiency during A&FC, Part 2 (ref 7, app A). The system was modified by HHI, prior to this test, by lowering the activation threshold from 93 to 89 percent engine power turbine speed (N_P). The system was evaluated at various power conditions ranging from autorotation to IRP. The 89 percent threshold was found to be adequate for providing an engine out warning within 4 to 7 seconds of a simulated partial power failure during low power descents (10 percent torque or less). At high power settings, significant N_R and accompanying N_P droop occurred with a simulated single engine failure. N_P decay below the activation threshold did not occur as frequently as reported in A&FC, Part 2. A one second collective control delay beyond the activation of the low main rotor RPM warning, following a simulated single engine failure at V_H resulted in N_R and N_P droop to 87.3 percent (para 54). N_P droop below the activation threshold of 89 percent will result in activation of the engine out warning light for the operating engine. This condition could be incorrectly interpreted by the pilot as a dual engine failure and, during terrain flight operations, could result in an immediate landing when it may have been possible to establish single engine flight. The possibility of a false indication of a dual engine failure in the event of a single engine failure, though less likely to occur with the reduced activation threshold, still remains a deficiency. The engine out warning system should be modified, prior to production, to insure that it provides a timely warning of engine failure under all flight conditions and does not activate unless an actual engine failure has occurred.

56. The engine out warning system was evaluated during this test to determine if the shortcoming reported in A&FC, Part 2, (ref 7, app A) had been corrected. Engine failures were simulated on the ground by turning the engine fuel switch off. The warning system operated consistently at 63 percent gas producer speed (N_g) for the No. 1 engine and 67.4 percent N_g for the No. 2 engine. The system operated correctly regardless of engine power lever position. Previously no engine out warning was available unless the engine power lever was in the fly position. The discrepancy between the activation thresholds for the No. 1 and No. 2 engines appeared to be a problem peculiar to the test aircraft (Equipment Performance Report (EPR) No. 80-17-3-8) and did not detract from the effectiveness of the system. The previously reported shortcoming of the engine out warning system not in accordance with design requirements has been corrected.

VIBRATION CHARACTERISTICS

57. The forward flight vibration characteristics of the YAH-64 were evaluated in both the 8-HELLFIRE and the 2-ferry tank (181 gallon) configurations. Tests were conducted at average density altitudes of 15,800 feet and 15,320 feet, respectively. Quantitative vibration data were gathered at the conditions shown in table 2 and are presented in figures 98 through 107, appendix E. This data is intended to supplement the data obtained during Parts 1 and 2 of this A&FC (ref 7, app A). Generally, the 4 per revolution (4/rev) vibratory accelerations, of 19.3 hertz, were most noticeable to the pilot but when below 0.2g were not objectionable (VRS 2-3). Other harmonic vibrations (1/rev, 2/rev and 8/rev) were not objectionable to either crewmember. The vibration characteristics observed in the 8-HELLFIRE configuration were essentially the same as those observed in the A&FC, Part 2. Though 4/rev vibration were slightly higher for the 2-ferry tank configuration they were not objectionable in level flight between the range of 50 and 91 KCAS (the maximum airspeed tested). A significant and objectionable increase in 4/rev vibration was noted, during IRP climbs at 80 to 90 KCAS, in this configuration. Vibration levels during climbs increased in both configurations but were most objectionable in the 2-ferry tank configuration (VRS 5-6). The vibrations observed during this evaluation were most apparent at the pilots station, as they have been during previous tests. The 4/rev vibration characteristics are essentially unchanged and remain objectionable during several flight regimes. As previously reported in Parts 1 and 2 of this A&FC program, the objectionable 4/rev vibration remains a shortcoming.

RELIABILITY AND MAINTAINABILITY

58. The reliability and maintainability features of the YAH-64 aircraft were evaluated throughout the test. Eleven EPR's were prepared and submitted during this evaluation and are listed in appendix F. This section is intended to summarize the most significant reliability and maintainability problems encountered.

59. Throughout the test program, accumulation of oil was observed on the deck below the main transmission, inside each engine cowling and dripping from the tail boom after accumulating in the maintenance access area aft of the main transmission. The majority of the oil appeared to be leaking from the engine nose gear box seals. A total of 90 ounces of oil was used throughout the test (87.9 flight hours) in the No. 1 engine nose gear box and 146 ounces in the No. 2 engine nose gear box. The excessive oil accumulation due to leakage of the engine nose gear boxes has been submitted as an EPR during A&FC, Part 2 (ref 7, app A) and is a previously reported shortcoming.

60. Throughout the evaluation, Heading and Attitude Reference System (HARS)/DASE interface and HARS alignment problems similar to those reported in A&FC, Part 2 (ref 7, app A) were encountered. The problem is not with the HARS units but with the special software program in the Back-Up Bus Controller unique to the test aircraft (USA S/N 77-23258). Three HARS changes were made throughout the test program when alignment problems were encountered. When these units were installed in another aircraft, which had a complete multiplex system to include a Fire Control Computer, the HARS units functioned normally. The HARS alignment and HARS/DASE interface problems, although annoying and responsible for maintenance down time, are not considered a problem area since they appear to be isolated to the test aircraft.

61. Numerous incidents of main transmission oil filter bypass buttons popping were encountered particularly at the high altitude test site at Bishop, California (EPR 80-17-3-4). The oil filter bypass buttons are designed to be activated by differential oil pressure caused by a clogged filter. Both left and right filter bowls were replaced to correct the problem when checks performed after several incidents showed the filters to be clean. A total of nine incidents of the buttons popping were reported and, on occassion, flights were delayed because of the problem. It appears that the differential pressure required to pop the filter button may be set too close to the operating pressure resulting in erroneous activation of the buttons. The erroneous activation of the main transmission oil filter bypass buttons is a shortcoming which should be corrected prior to operational use of the helicopter.

62. Approximately two test days were lost throughout the program for unscheduled maintenance required due to delamination of the main rotor blade doublers along the trailing edge. This problem occurred on all four blades on one flight (EPR 80-17-3-10) and on one blade on another flight (EPR 80-17-3-9). A third incident occurred between the two reported for this test during a flight conducted by HHI. Following the third incident where all four main rotor blades delaminated on one flight an engineering change was issued to cover the main rotor blade doublers with a layer of fiberglass. Subsequent to the engineering change, no further incidents of blade delamination occurred.

63. Throughout the test program, replacement of the stabilator pivot bushings both in the stabilator and vertical fin were responsible for the majority of unprogrammed maintenance down time. On eleven reported occasions, the bushings broke loose requiring replacement and on at least two of these occasions after only one flight since installation (EPR 80-17-3-5). Prior to the end of this test, a stabilator with a modified pivot bushing system (P/N 7-211122601) was installed to correct this problem. A total of twelve flights were flown on the stabilator and new bushings, with no reoccurrence of the problem. The new stabilator pivot bushing installation appears to have corrected the problem of recurring stabilator bushing failures.

64. In order to remove a previously reported shortcoming, small doors were installed on the engine cowlings to allow engine oil levels to be checked without opening the cowlings. The doors were installed to open into the relative airflow and required a screw driver to open the latch. Should the latch fail in flight, the doors could be blown open by the airflow and torn from the aircraft. The previously reported shortcoming of the difficulty in accurately determining the engine oil levels without opening the engine cowlings has been corrected. The engine oil level inspection doors should be modified to streamline into the wind and a quick release latch, which requires no special tool to open, should be incorporated.

SUBSYSTEM TESTS

Engine Starts

65. Engine starting characteristics were evaluated throughout the test program. Tests included an evaluation of normal starts on the ground, inflight engine restarts, and engine restarts after a 10 to 20 minute heat soak-back period on the ground. Rotor brake locked starts were also performed. Engine cross-bleed starts

were not evaluated due to failure of a shaft driven compressor (SDC) shutoff valve installed to prevent SDC air from entering the pressurized air system thus allowing the No. 2 engine to be started using bleed air from the No. 1 engine. The SDC shutoff valve was installed only for HHI engine cross-bleed start tests and is not part of the production system. The engine start procedures used during this test were those prescribed in the operator's manual (ref 12, app A) as modified by the airworthiness release (ref 10).

66. Normal engine starts were performed on the ground at the test site elevations of 800 feet, 4120 feet, and 9980 feet. Tests were conducted using either the SDC or an external air source to provide pressurized air for starting the engines. Representative time history plots of normal engine starts are provided as figures 108 and 109, appendix E. All engine starts were successful and are satisfactory.

67. Engine starts, on the ground following a 10 to 20 minute heat soak-back period, were conducted at test site elevations of 800 feet, 4120 feet, and 9980 feet. Tests were conducted following a normal engine shutdown and a minimum of a 10 minute delay before attempting an engine restart. The engine power lever was not advanced to the idle position until TGT had decreased to 275°C. Time history plots of typical engine starts, under these conditions, are shown in figures 110 and 111, appendix E. All start attempts were successful and essentially the same as normal starts with the exception of slightly higher TGT indications. Engine starts following a 10 to 20 minute heat soak-back period were satisfactory.

68. Inflight engine starts were performed primarily to evaluate the high altitude restart capability of the YT700-GE-701 engines and to evaluate specification compliance. Tests were conducted at density altitudes up to 17,720 feet, sideslip angles ranging from 0 to ±15 degrees in level flight, descents and autorotation. Test conditions are shown in table 2. The engine power lever was not advanced to the idle position until TGT had decreased to 150°C. A time history of a typical high altitude engine restart is shown in figure 112, appendix E. All engine start attempts were successful and no adverse conditions were noted. Engine restarts could not be evaluated for specification compliance since the aircraft was restricted to a maximum density altitude of 20,000 feet (ref 10, app A) and level flight could not be maintained with one engine inoperative at this altitude. The inflight restart capability of the YT700-GE-701 engines, up to a density altitude of approximately 18,000 feet, is satisfactory.

Engine Fuel Suction Feed System

69. Tests were conducted to determine the adequacy of the engine fuel suction feed system during high altitude/hot day conditions using JP-4 fuel. A total of 22 flights were conducted during July 1982 at Yuma Proving Ground, Arizona. Test conditions are presented in table 4. Fuel temperature measurements were taken immediately upon completion of each flight. A comparison of data obtained during these tests with the suction feed envelope requirements of the system specification (ref 11, app A) is shown in figure 113, appendix E. Illumination of the engine fuel pressure warning lights was observed at pressure altitudes of 13,820 feet and 14,980 feet with fuel temperatures of 91 and 94°F, respectively. The engine fuel pressure warning lights are designed to illuminate when fuel pressure drops below 9 ±1 pounds per square inch, gauge (psig) at the fuel filter. During normal operation, fuel pressure is between 45 and 90 psig at the fuel filter. Illumination of a fuel pressure warning light indicates abnormal operation and emergency procedures published in the operator's manual (ref 12, app A) require activation of the fuel boost pump. Operation with JP-5 fuel was satisfactory at a pressure altitude of 16,100 feet and a fuel temperature of 107°F. JP-5 fuel should be used for operation above pressure altitudes of 13,000 feet with fuel temperatures of 90°F or greater. The engine fuel suction feed system failed to meet the JP-4 fuel suction feed requirements of 20,000 feet pressure altitude at a fuel temperature of 95°F as specified in paragraph 3.7.5.7.p of reference 11, in that illumination of the engine fuel low pressure warning light was observed at a pressure altitude of 13,820 feet with a fuel temperature of 91°F.

EXTERNAL NOISE SURVEY

70. An external noise survey was conducted under the research and development contract requirements. Acoustical measurements were taken in level flight, descents and hover. Support for this test was provided by NASA/Ames Research Center using a specially equipped YO-3A aircraft for inflight noise measurements. Test results will be published as a separate report by the US Army Aeromechanics Laboratory, Moffett Field, California.

MISCELLANEOUS

71. Testing was accomplished to determine the status of shortcomings previously reported during Parts 1 and 2 of the A&FC (refs 6 and 7, app A). As a result of this evaluation, fifteen short-

Table 4. JP-4 Fuel Tests[1]

Date	Fuel Temp (°F)	Pressure Altitude (ft)	OAT (°C)	Fuel Remaining (fwd/aft)
1 July	84	3420	22	430/80
2 July	80	1960	28	440/400
2 July	84	4980	21	790/400
2 July	94	14,980	3	440/130[2]
6 July	82	6560	18	759/510
7 July	80	9500	10	410/70
7 July	82	9500	10	380/80
8 July	82	2220	28	330/80
8 July	92	12,480	7	470/260
9 July	84	2040	29	370/80
12 July	100	6700	22	400/180
12 July	90	1440	31	500/100
13 July	96	11,300	11	470/80
14 July	92	5020	28	350/100
15 July	104	3680	29	810/580
16 July	92	12,160	7	410/170
16 July	99	2380	27	600/400
19 July	92	6200	24	360/130
20 July	92	960	30	310/320
21 July	91	13,820	4	860/400[2]
22 July	102	3120	27	930/210
23 July	93	3360	29	360/400

NOTES:

[1]JP-4 Fuel, Reid Vapor Pressure of 2.6 pounds per square inch absolute (psia)
[2]Engine fuel pressure warning light observed during flight

comings remain valid and are listed below. The previously reported shortcomings are listed in order of relative importance. The AH-64 Program Manager and HHI have identified corrective action for the shortcomings shown with asterisks (*) (ref 18, app A).

a. Failure of the Environment Control Unit to provide adequate crew station cooling.*

b. The objectionable 4/rev (19.3 Hz) vertical vibration characteristics.

c. The restriction to the pilot's field of view caused by window edge distortion, the overhead circuit breaker panel, canopy reflection, CPG helmet, and the PNVS turret.

d. The poor design of the pilot's fuel control panel.*

e. The lack of a reliable indication of parking brake status.

f. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls.

g. The poor location of the pilot engine control quadrant.*

h. The poor design of the collective pitch control friction mechanism.*

i. The excessive accumulation of oil on the main transmission deck and in the upper fairing maintenance access area.*

j. The poor location of the tail wheel unlock light.

k. The high inherent friction of the engine power control levers.

l. The illumination of the Auxiliary Power Unit (APU) ON advisory light prior to the APU stabilizing at 100 percent rpm.*

m. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight.*

n. The poor anthropometric design of the pilots cyclic grip.*

o. The constant illumination of the lower green segment light on the Marconi vertical scale.

72. The previously reported shortcoming of the Marconi instruments failing to display the full green range during normal operation has been corrected.

CONCLUSIONS

GENERAL

73. Based on the A&FC flight test of the YAH-64 helicopter, the following conclusions were reached:

a. The YT700-GE-701 engines provided a significant increase in power available, compared to the YT700-GE-700R engines, and proved to be very reliable throughout the test program (para 39).

b. The hover, vertical climb, and level flight performance is significantly improved by the YT 700-GE-701 engines (para 7).

c. The slope landing characteristics on lateral slopes of up to 9 degrees and longitudinal slopes of up to 10 degrees are satisfactory (paras 36 and 37).

d. The gust response in hover and forward flight is satisfactory with the production DASE software (paras 49 and 50).

e. Aircraft response to simulated DASE failures in hover and forward flight, is now satisfactory (paras 51 and 52).

f. The new stabilator pivot bushing installation (Part No. 7-211122601) appears to have corrected the problem of recurring stabilator bushing failures (para 63).

g. The inflight restart capability of the YT700-GE-701 engines up to a density altitude of approximately 18,000 feet, is satisfactory (para 68).

h. One enhancing characteristic, the automatic contingency power feature of the YT700-GE-701 engine, was identified (para 40).

i. The previously reported deficiency, the possibility of a false indication of a dual engine failure in the event of a single engine failure, has not been corrected (para 55).

j. Five previously reported shortcomings have been corrected (paras 50, 53, 56, 64, and 72).

k. Fifteen previously reported shortcomings remain (para 71).

l. Three additional shortcomings have been identified (paras 41, 45, and 61).

m. Eleven equipment performance reports were submitted during this test (para 58).

n. Nine incidents of specification noncompliance were noted, three of which were considered acceptable (para 77).

ENHANCING CHARACTERISTIC

74. The automatic contingency power feature of the YT 700-GE-701 engine (para 40).

DEFICIENCY

75. The following deficiency was initially identified during A&FC, Part 2 and upon re-evaluation has not yet been adequately corrected (see app D for definition of deficiency used in this report).

The possibility of a false indication of a dual engine failure in the event of a single engine failure (para 55).

SHORTCOMINGS

76. The following shortcomings (listed in order of relative importance) have been identified. Those shown with asterisks (*) were reported on previous USAAEFA tests and still exist (see app D for definition of shortcoming used in this report).

a. The undesireable engine/airframe response during power application from a zero torque condition (para 41).

b. Failure of the Environment Control Unit to provide adequate crew station cooling.*

c. The objectionable 4/rev (19.3 Hz) vertical vibration characteristics.*

d. The restriction to the pilot's field of view caused by window edge distortion, the overhead circuit breaker panel, canopy reflection, CPG helmet, and the PNVS turret.*

e. The poor design of the pilot's fuel control panel.*

f. The excessively large longitudinal cyclic inputs required when retrimming during IMC flight with attitude hold ON (para 45).

g. The lack of a reliable indication of parking brake status.

h. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls.*

i. The poor location of the pilot engine control quadrant.*

j. The poor design of the collective pitch control friction mechanism.*

k. The erroneous activation of the main transmission oil filter bypass buttons (para 61).

l The excessive accumulation of oil on the main transmission deck and in the upper fairing maintenance access area.*

m. The poor location of the tail wheel unlock light.*

n. The high inherent friction of the engine power control levers.*

o. The illumination of the Auxiliary Power Unit (APU) ON advisory light prior to the APU stabilizing at 100 percent rpm.*

p. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight.*

q. The poor anthropometric design of the pilots cyclic grip.*

r. The constant illumination of the lower green segment light on the Marconi vertical scale.*

SPECIFICATION COMPLIANCE

77. The YAH-64 was found to be in noncompliance with the following paragraphs of the System Specification for the AH-64A Advanced Attack Helicopter DRC-S-H10000B. Additional specification noncompliance, beyond the scope of this evaluation may exist. Paragraphs preceded by asteristics (*), although in noncompliance, are considered acceptable.

*a. 10.3.4.1 - Variation of longitudinal control position with airspeed was neutral for a level flight trim airspeed of 60 KCAS (para 26).

*b. 10.3.4.4.2 - Average response time to a longitudinal control step input was less than 0.7 sec by 0.2 sec (para 32).

*c. 10.3.5.2.1 - Average response time to a lateral control step input was less than 0.7 sec by 0.3 sec (para 33).

d. 10.3.10.6 - Landings on 15 degree lateral slopes were not possible due to large fuselage roll attitudes and the possibility of tire/rim separation during operation on a lateral slope of 10-1/2 degrees (para 36).

e. 10.3.10.6 - Landings on a 12 degree longitudinal slope were not satisfactory as the main rotor droop stops were damaged when landing on a 12 degree longitudinal slope (para 37).

f. 10.3.2.5 - The cyclic control would not maintain the zero force position selected by the pilot (para 46).

g. 10.3.8.1.1 - The available collective control delay time at V_H was less than 2.0 sec by 0.8 sec (para 54).

h. 3.7.5.7.p - Engine fuel suction feed system did not meet the JP-4 fuel suction feed envelope of 20,000 feet presure altitude at a fuel temperature of 95°F in that illumination of the engine fuel low pressure warning light was observed at a pressure altitude of 13,820 feet with a fuel temperature of 91°F (para 69).

RECOMMENDATIONS

78. The following recommendations are made:

a. Correct the deficiency prior to operational use (para 75).

b. Correct the shortcomings prior to operational use (para 76).

c. Due to unusable airspeed indications, takeoff performance data for climbout airspeeds below 35 KTAS should not be presented in chapter 7 of the operator's manual (para 11).

d. The procedure for level acceleration takeoffs, specified in paragraph 8-50 of the operator's manual, should be changed to read as indicated (para 11).

e. Place the following CAUTION in chapter 2 of the operator's manual (para 34):

CAUTION

> Do not rely on the position of the parking brake handle as an indication of parking brake status. Brakes must be reset or released, as appropriate, to determine correct status of parking brake.

f. The lateral slope landing envelope should be limited to 9 degrees maximum slope angle and 12 degrees fuselage roll attitude (para 36).

g. Place the following CAUTION in chapter 8 of the operator's manual (para 36):

CAUTION

> During slope landing operations, use the primary attitude indicator to insure that fuselage roll attitude does not exceed 12 degrees when centering the cyclic. Should fuselage roll attitude reach 12 degrees before the cyclic has been centered, reposition the aircraft to a shallower slope.

h. The longitudinal slope landing envelope should be limited to 10 degrees maximum slope angle (para 37).

i. The requirement for a slope landing capability on greater than 10 degree longitudinal slopes and 9 degree lateral slopes should be re-evaluated by the user (paras 36 and 37).

j. The engine response to variations in main rotor speed should be further investigated (para 41).

k. The engine oil level inspection doors should be modified to streamline into the wind and a quick release latch, which requires no tool to open, should be incorporated (para 63).

l. JP-5 fuel should be used for operation above pressure altitudes of 13,000 feet with fuel temperatures of 90°F or greater (para 68).

APPENDIX A. REFERENCES

1. Final Report, USAAEFA Project No. 74-07-2, *Development Test 1, Advanced Attack Helicopter Competitive Evaluation, Hughes YAH-64 Helicopter*, December 1976.

2. Final Report, USAAEFA Project No. 77-36, *Engineer Design Test 1, Hughes YAH-64 Advanced Attack Helicopter*, September 1978.

3. Final Report, USAAEFA Project No. 78-23, *Engineer Design Test 2, Hughes YAH-64, Advanced Attack Helicopter*, June 1979.

4. Final Report, USAAEFA Project No. 80-03, *Engineer Design Test 4, YAH-64 Advanced Attack Helicopter*, January 1980.

5. Final Report, AVNDTA, *Engineer Design Test, Government (EDT-G) 5 of the Advanced Attack Helicopter (AAH) (YAH-64)*, May 1981, revised June 1981.

6. Final Report, USAAEFA Project No. 80-17-1, *Airworthiness and Flight Characteristics Test, Part 1, YAH-64 Advanced Attack Helicopter*, September 1981.

7. Final Report, USAAEFA Project No. 80-17-2, *Airworthiness and Flight Characteristics Test, Part 2, YAH-64 Advanced Attack Helicopter*, February 1982.

8. Letter, AVRADCOM, DRDAV-D, 8 March 1981, subject: Airworthiness and Flight Characteristics (A&FC) Test of YAH-64 Advanced Attack Helicopter, Prototype Qualification Test-Government (PQT-G), Part 3 and Production Validation Test-Government (PVT-G) for Handbook Verification.

9. Test Plan, USAAEFA Project No. 80-17-3, *Airworthiness and Flight Characteristics (A&FC) Test of YAH-64 Advanced Attack Helicopter, Prototype Qualification Test-Government (PQT-G), Part 3 and Production Validation Test-Government (PVT-G) for Handbook Verification*, January 1982.

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12. Draft Technical Manual, TM55-1520-238-10, *Operator's Manual for Army YAH-64 Helicopter,* 1 May 1981.

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15. Technical Manual, TM-5-530, *Materials Testing,* 6 February 1971.

16. Letter, AVRADCOM, DRDAV-DP, 2 August 1982, subject: Request for Installed Engine (T700-GE-701) Performance Data.

17. Letter, Hughes Helicopters Incorporated, Mr. John Engle, 9 July 1982, subject: DASE Production Software.

18. Status Meeting (previously reported YAH-64 deficiencies and shortcomings) Hughes Helicopters Incorporated, YAH-64 Program Manager, Development and Qualification Directorate (AVRADCOM), USAAEFA, St. Louis, Mo., 15 January 1982.

19. Airman's Information Manual, *Basic Flight Information and ATC Procedures,* Federal Aviation Administration, U.S. Department of Transportation, published quarterly.

APPENDIX B. DESCRIPTION

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GENERAL

1. The YAH-64 Advanced Attack Helicopter (fig. 1) is a tandem, two-place twin turbine engine, single main rotor aircraft manufactured by Hughes Helicopters Incorporated (HHI). The main rotor is a four-bladed fully articulated system supported by a stationary mast which transmits flight loads directly to the fuselage. The tail rotor is a four-bladed, semi-rigid, delta-hinged system incorporating elastromeric teetering bearings. The rotors are driven by two General Electric YT700-GE-701 engines through the power train shown in figure 2. An AiResearch GTCP 36-55(c) auxiliary power unit (APU), is installed to drive the accessory section of the main transmission when the rotors are not turning. This provides pneumatic power for engine starting as well as electrical and hydraulic power for aircraft systems. The aircraft is designed to carry ordinance stores internally in the ammunition bay and externally on the four wing store positions. The YAH-64 is designed to operate during day, night and marginal weather combat conditions using the Martin-Marietta Target Acquisition Designation System (TADS)/Pilot Night Vision Sensor (PNVS). The test aircraft, S/N 77-23258, was configured with an aerodynamic mockup of the TADS/PNVS, 30mm chain gun and eight HELLFIRE missiles to represent the primary mission configuration (photos 1 through 8). Additional HELLFIRE missiles, 2.75-inch rocket launchers, and external fuel tanks were attached to or removed from the wing pylons, as appropriate, to attain each of the required test configurations (photos 9 through 18). The major modifications and external configuration changes since A&FC, Part 2 and the major external difference between the test aircraft and the production aircraft configuration are presented in in table 1. Drag estimates provided by HHI are included in this table for the items not on the test aircraft but described in the System Specification for the program and for any external test instrumentation equipment. The Back Up Control System (BUCS) was not operational during this test.

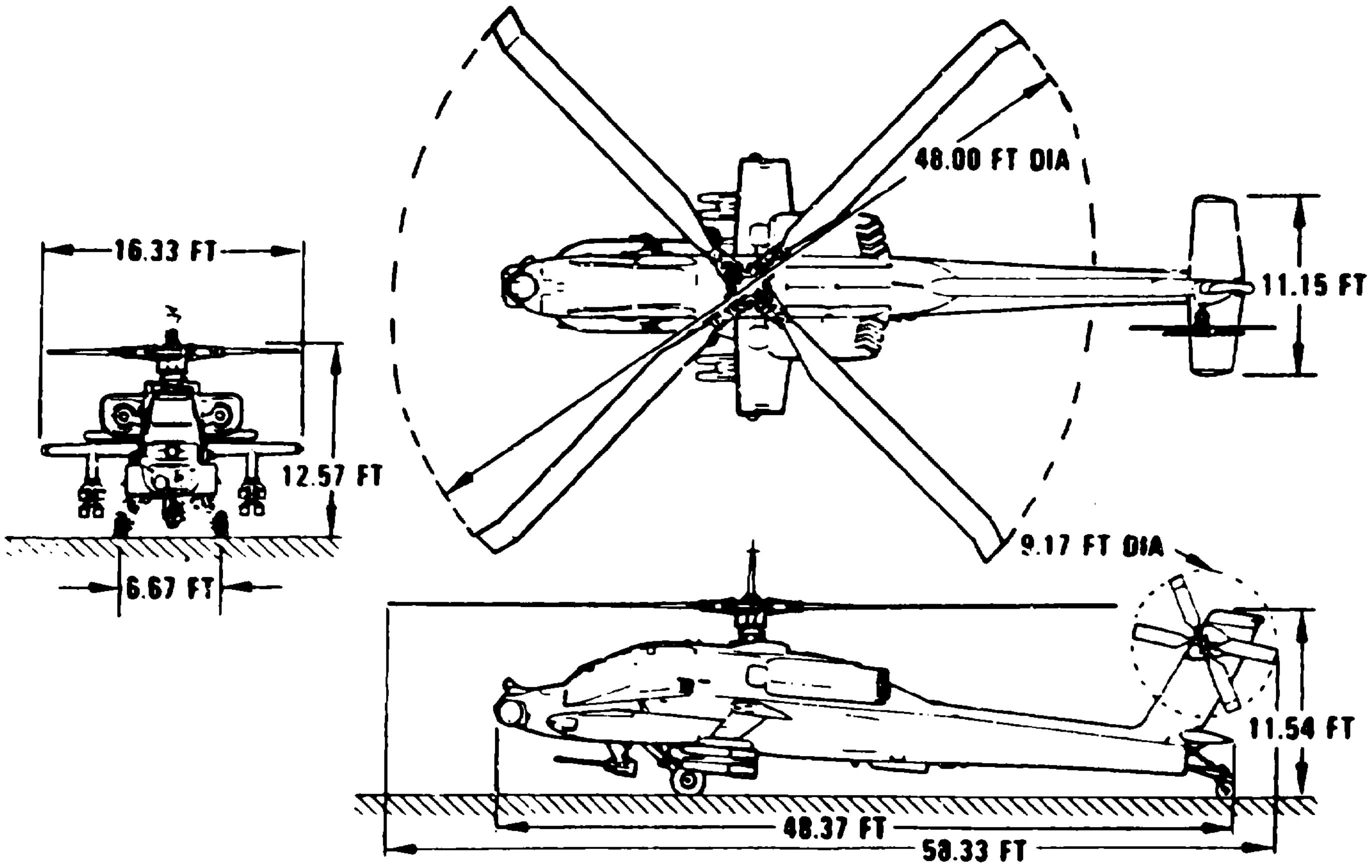


Figure 1. Aircraft Dimensions

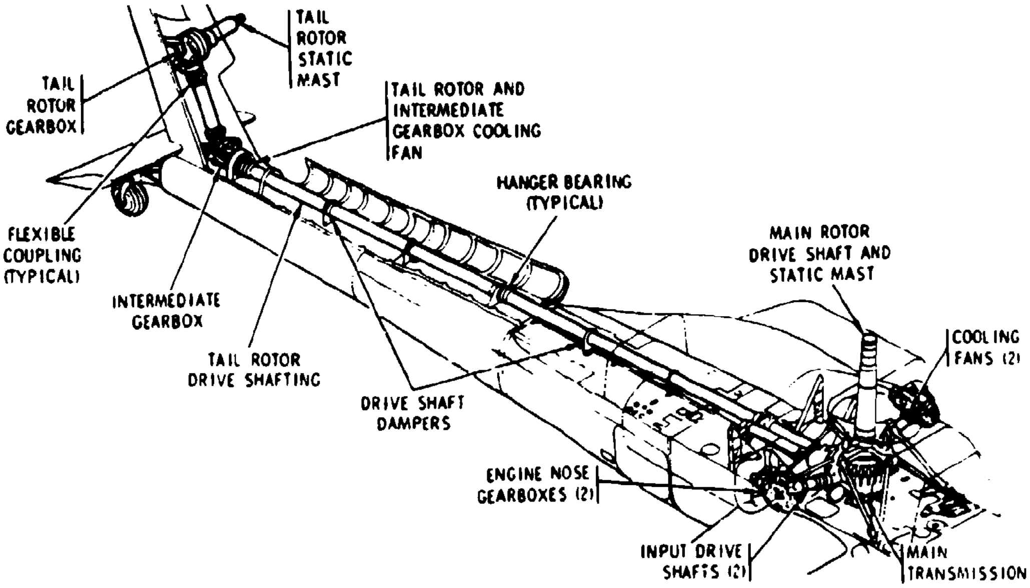


Figure 2. Powertrain

Photo 1. Front View (8-HELLFIRE Configuration)

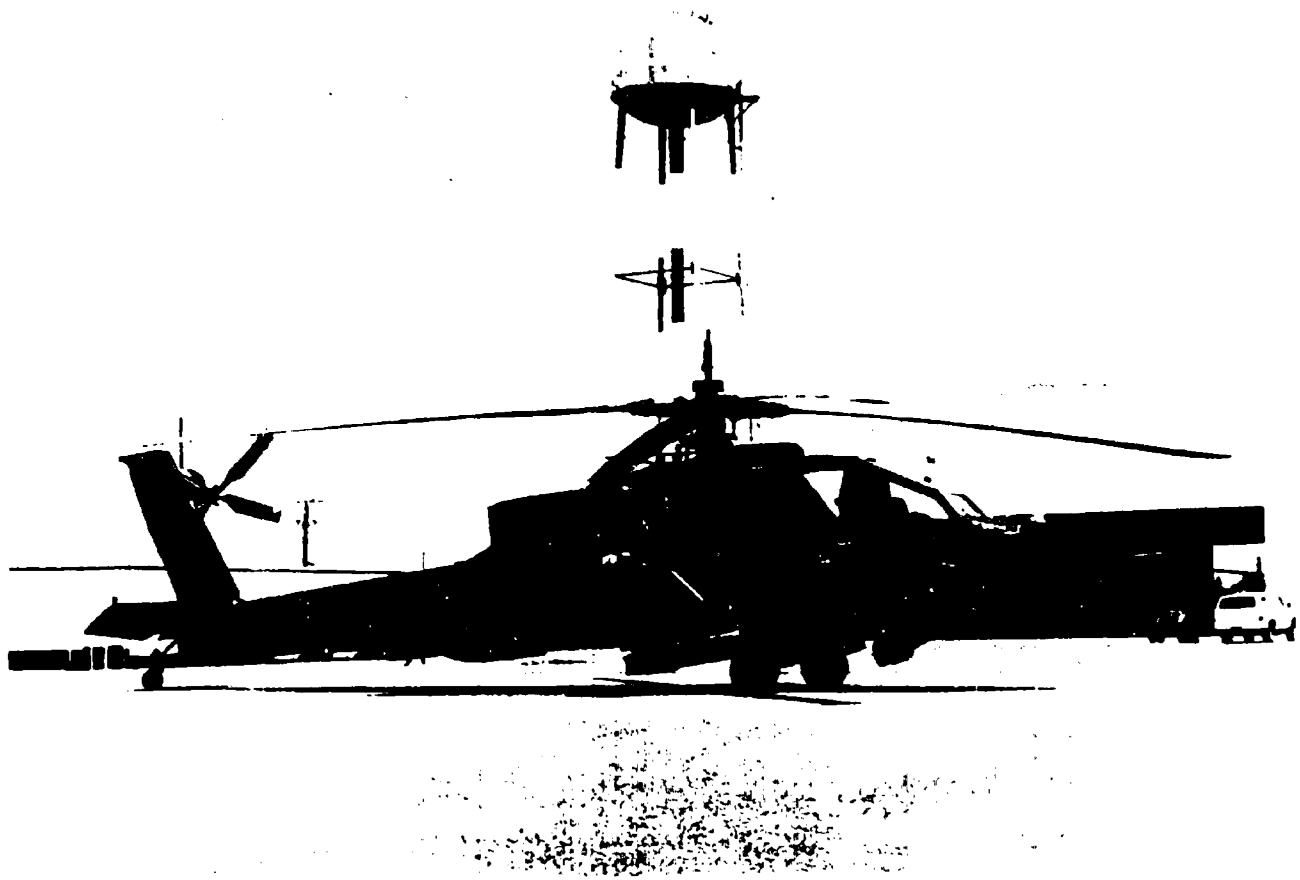


Photo 2. Right Front Quartering View (8-HELLFIRE Configuration)

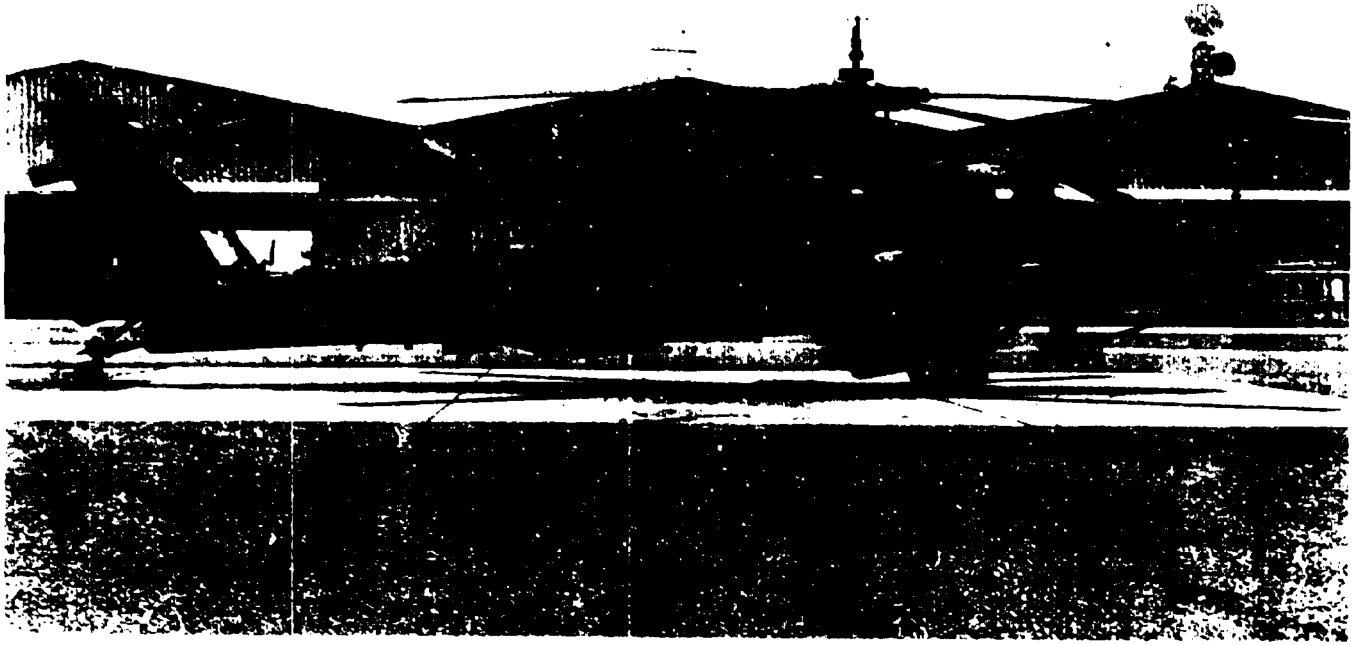


Photo 3. Right View (8-HELLFIRE Configuration)

Photo 4. Right Rear Quartering View (8-HELLFIRE Configuration)

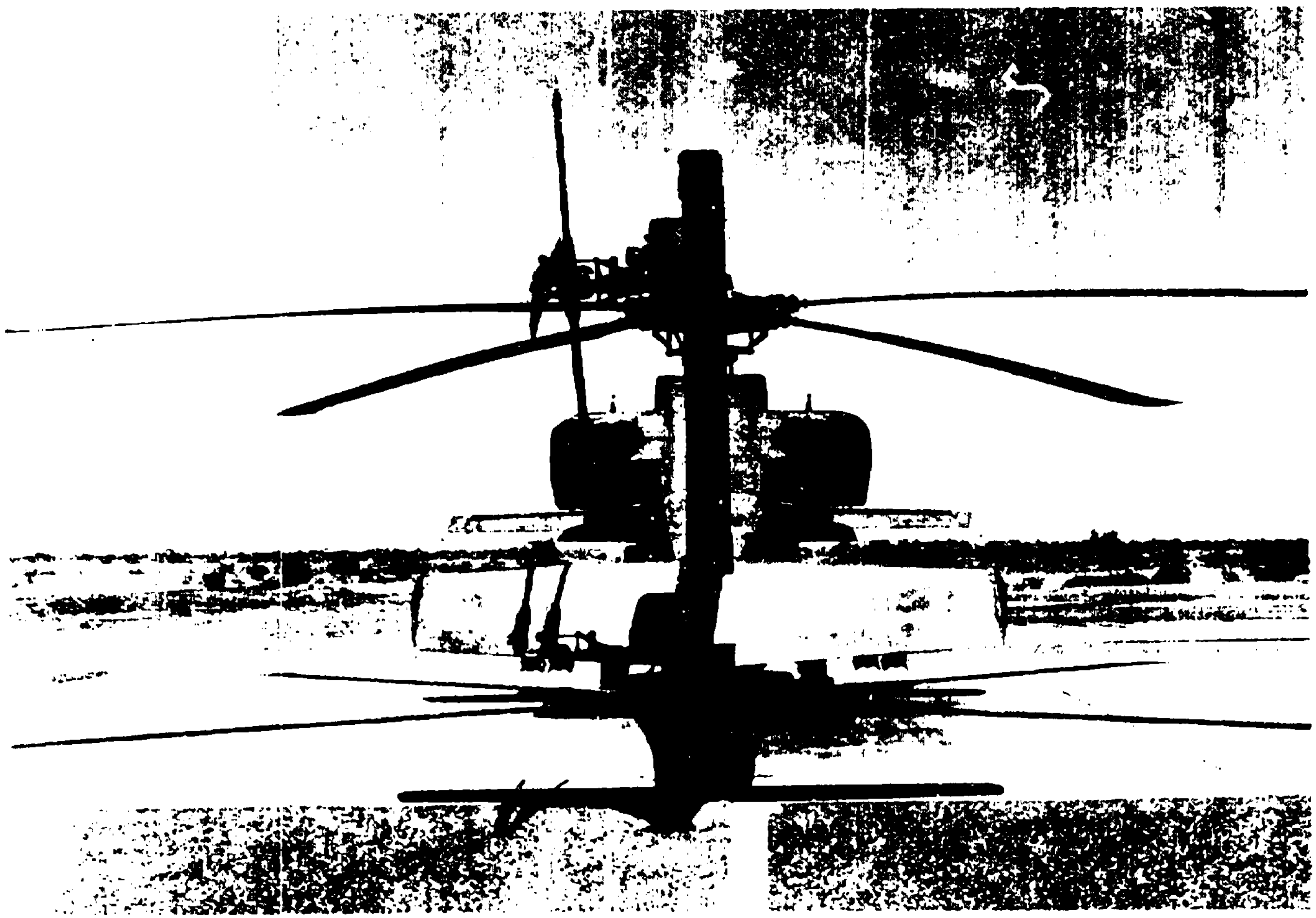


Photo 5. Rear View (8-HELLFIRE Configuration)

Photo 6. Left Rear Quartering View (8-HELLFIRE Configuration)

Photo 7. Left View (8-Hellfire Configuration)

Photo 8. Left Front Quartering View (8-HELLFIRE Configuration)

Photo 9. Front View (16-HELLFIRE Configuration)

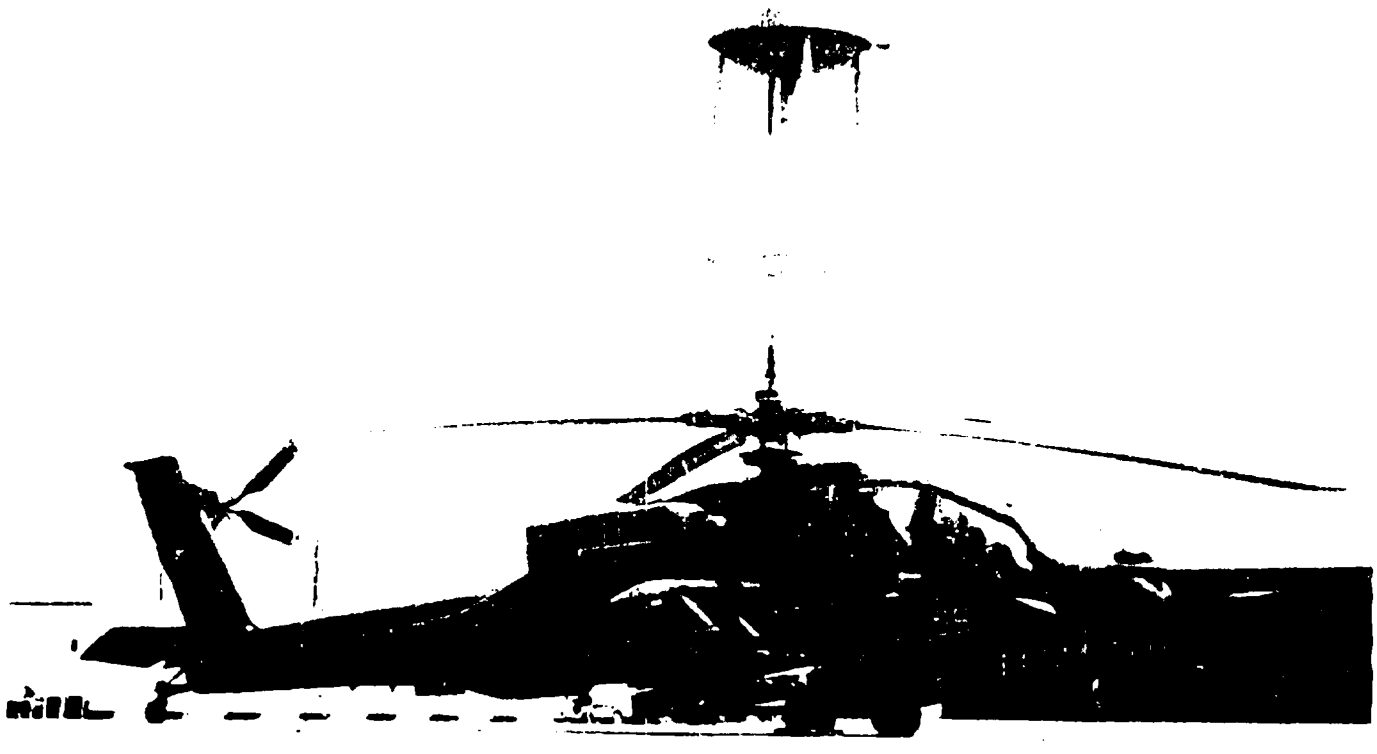


Photo 10. Right Front Quartering View (16-HELLFIRE Configuration)

Photo 11. Front View (Clean Configuration)

Photo 12. Right Front Quartering View (Clean Configuration)

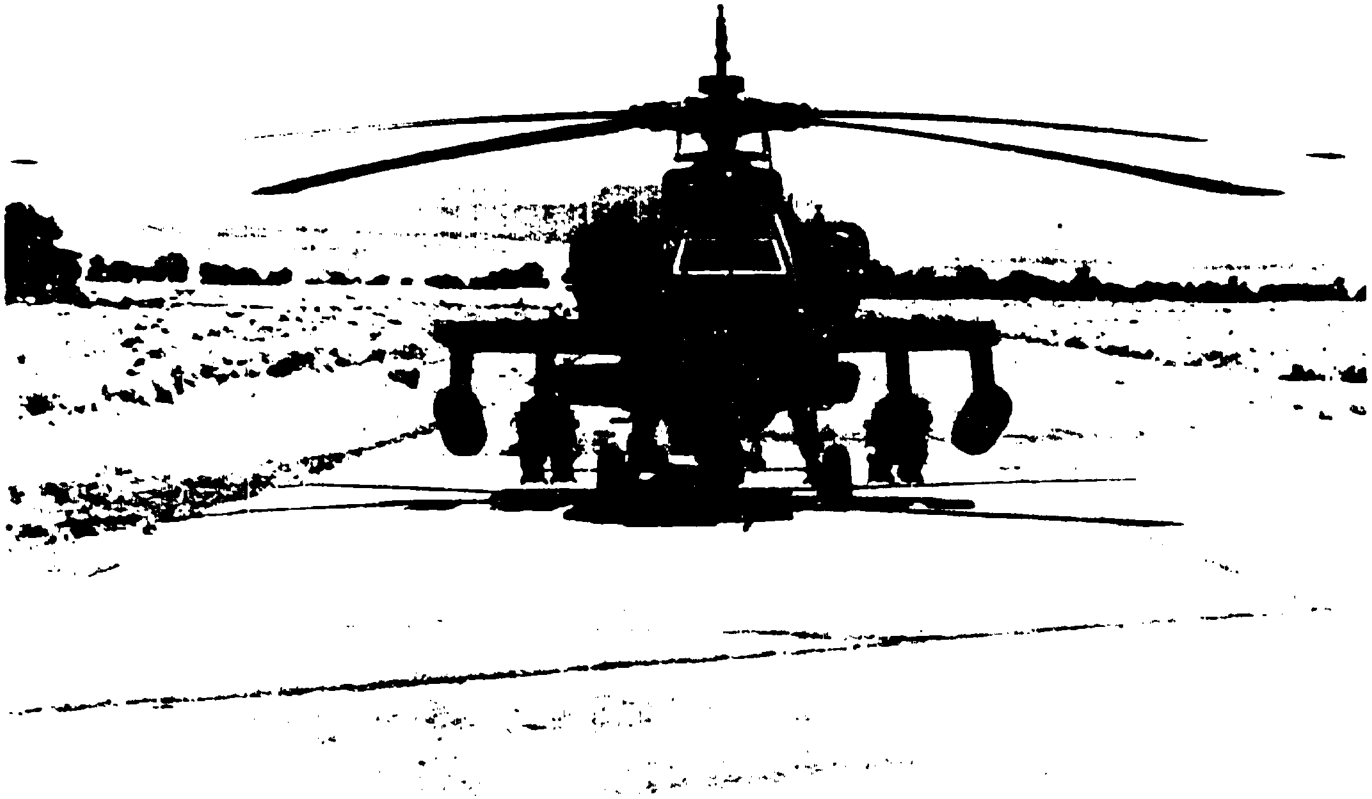


Photo 13. Front View (8-HELLFIRE, 2-2.75 in. Rocket Launcher Configuration)

Photo 14. Front Front Quartering View
(8-HELLFIRE, 2-2.75 in. Rocket Launcher Configuration)

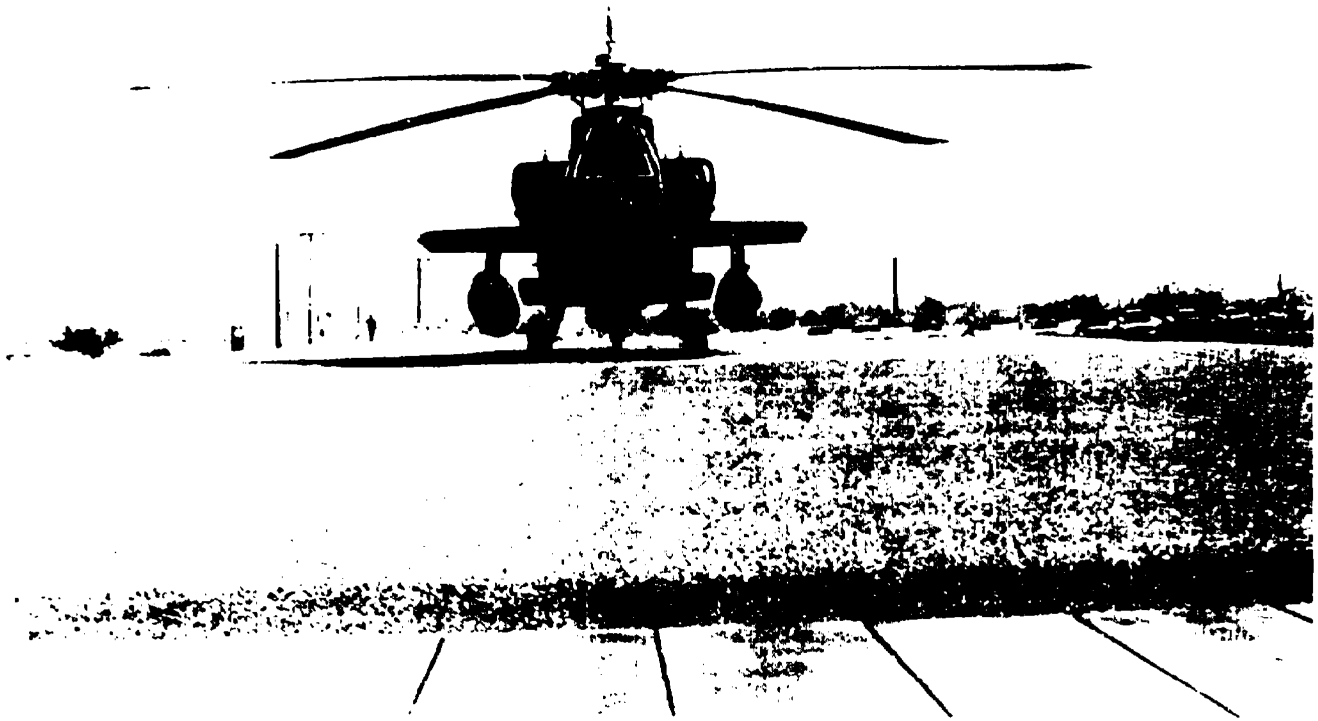


Photo 15. Front View (2-Ferry Tank (181-Gallon) Configuration)

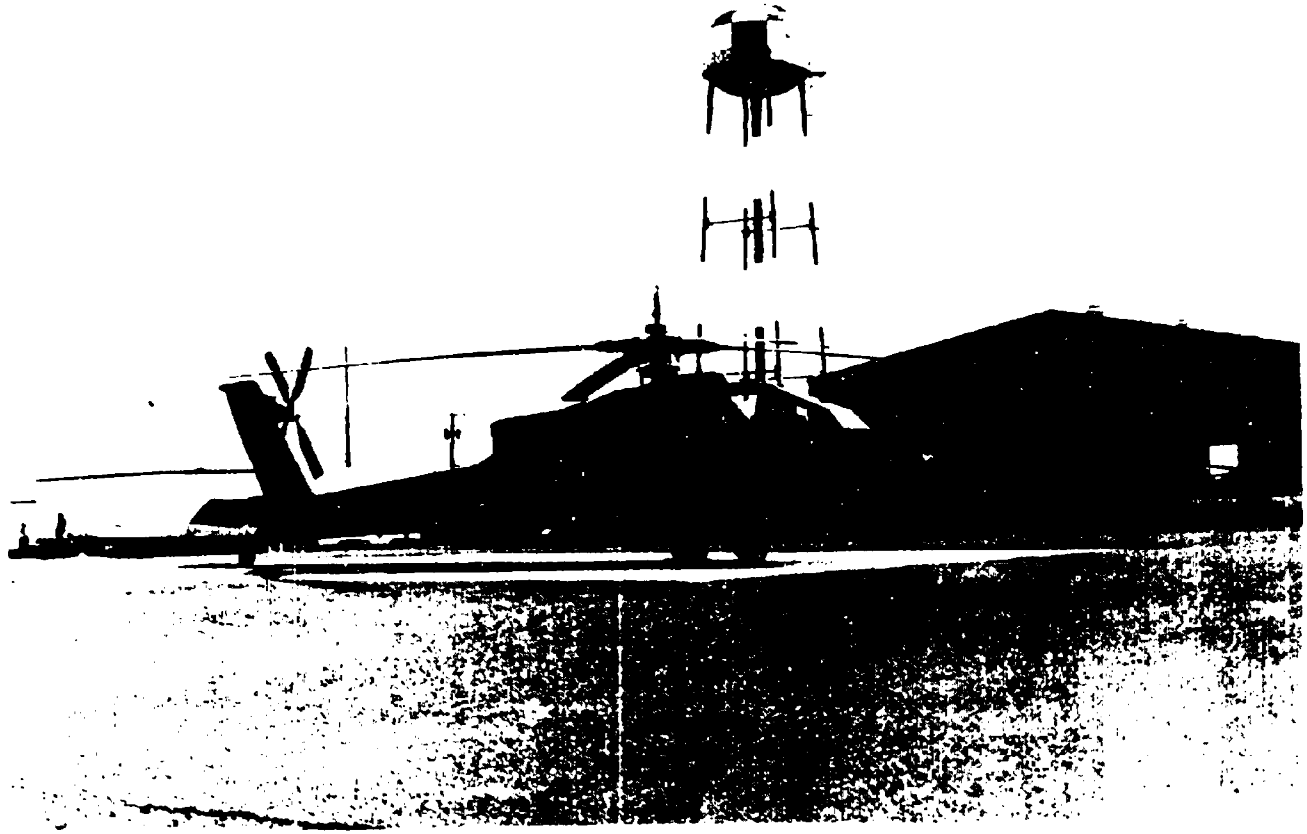


Photo 16. Right Front Quartering View
(2-Ferry Tank (181-Gallon) Configuration)

Photo 17. Front View (8-HELLFIRE Lateral Offset Configuration)

Photo 18. Right Front Quartering View
(8-HELLFIRE Lateral Offset Configuration)

Table 1. Configuration of Test Aircraft (USA S/N 77-23258)

Configuration changes since A&FC Part 2:

a. Re-installed M/R instrumentation rotating Pulse Code Modulation cannister.

b. Installed YT700-GE-701 calibrated engines (G02J Electrical Control Units)

c. Modified engine Infrared (IR) Suppressor nozzles and added support brackets.

d. Installed engine oil level inspection doors in engine cowling.

e. Installed Digital Automatic Stabilization Equipment (DASE) production software (Program No. 1571 DASE-001 REVNA).

Items not on the test aircraft but defined in the system specification for the production program:

	Estimated drag ΔF_e (ft^2)
a. Maintenance step (left side of aircraft)	0.06
b. 30mm ammunition feed chute	0.60

Items not in the system specification but installed on the test aircraft:

	Estimated drag ΔF_e (ft^2)
a. Airspeed boom with angle of attack and sideslip vanes	0.90
b. Instrumentation canister located on main rotor hub	0.14
c. Telemetry antenna brackets located on scissors assembly and telemetry antenna and bracket located on underside of tail cone transition section	0.20
d. Instrumented main and tail rotor pitch change links	0.01

DIMENSIONS AND GENERAL DATA

Main Rotor

Diameter (ft)	48
Blade chord (in.)	21.0*
Main rotor total blade area (ft^2)	166.5
Main rotor disc area (ft^2)	1809.56
Main rotor solidity (thrust weighted, no tip loss)	0.092
Airfoil	HH-02**
Twist (deg)	-9
Number of blades	4
Rotor speed at 100 percent N_R (RPM)	289.3
Tip speed at 100 percent N_R (ft/sec)	727.09
Gear ratio (engine to main rotor)	72.424322

Tail Rotor

Diameter (ft)	9.17
Chord; constant (in.)	10
Tail rotor total blade area (ft^2)	14.89
Tail rotor disc area (ft^2)	66.0
Tail rotor solidity	0.2256
Airfoil	NACA 632-414 (modified)
Twist (deg)	-8.8
Number of blades	4
Rotor speed at 100 percent N_R (RPM)	1403.4
Distance from main rotor mast centerline (C_L) (ft)	29.67
Tip speed at 100 percent N_R (ft/sec)	673
Teetering angle (deg)	35
Maximum blade angle (deg)	27

Horizontal Stabilator

Weight (lb)	77.3
Area (ft^2)	33.36
Span (ft)	11.15
Tip chord (ft)	2.65
Root chord (ft)	3.60
Airfoil	NACA 0018
Geometric aspect ratio	3.41

*Includes tips
**Outer 20 inches swept 20 degrees and transitioned to an NACA 006 airfoil

Horizontal Stabilator (continued)

Incidence of chord line (deg)	Variable (45 degrees leading edge up to 10 degrees leading edge down).
Sweep of leading edge (deg)	2.89
Sweep of trailing edge (deg)	-7.23
Dihedral (deg)	0

Vertical Stabilizer

Area (from boom C_L) (ft^2)	32.2
Span (from boom C_L) (in)	113.0
Root chord (at boom C_L) (in)	44.0
Geometric aspect ratio	2.5
Airfoil	NACA 4415 (modified)
Sweep of Leading edge (deg)	29.4
Vertical stabilizer trailing edge deflection	16 deg left above W.L. 196.0

Wing

Span (ft)	16.33
Mean aerodynamic chord (in.)	45.9
Total area (ft^2)	61.59
Flap area (ft^2)	8.71 (fixed)
Airfoil at root	NACA 4418

Aircraft

Fuel quantity (gals.)	369
Design gross weight (lb)	14694
Maximum gross weight (lb)	17850

FLIGHT CONTROL DESCRIPTION

General

2. The YAH-64 helicopter employs a single hydromechanical irreversible flight control system. The hydromechanical system is mechanically activated with conventional cyclic, collective and directional pedal controls, through a series of push-pull tubes attached to four airframe mounted hydraulic servoactuators. The four hydraulic servoactuators control longitudinal cyclic, lateral cyclic, main rotor collective and tail rotor collective pitch. Cyclic and directional servoactuators incorporate integral

stability augmentation system (SAS) actuators. Hydraulic power is supplied by two independent 3000-psi hydraulic systems which are powered by hydraulic pumps mounted on the accessory gearbox of the main transmission to allow full operation under a dual-engine failure condition. A Digital Automatic Stabilization Equipment (DASE) system is installed to provide rate damping and command augmentation. The DASE is limited to ± 10 percent of control authority in roll and yaw. The longitudinal cyclic hydraulic servoactutor allows 20 percent forward and 10 percent aft control authority in the pitch axis. The DASE also provides attitude hold and a Hover Augmentation System (HAS). An electrically-actuated horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. Movement of the stabilator can be controlled either manually or automatically. A trim feel system is incorporated in the cyclic and directional controls to provide a control force gradient with control displacement from a selected trim position. A trim release switch, located on the cyclic grip, provides momentary interruption of the force gradient in all axes simultaneously to allow the cyclic or pedal controls to be placed in a new reference trim position. Full control travel is 10.2 inches in the longitudinal control, 9.2 inches in the lateral control, 11.5 inches in the collective control, and 4.3 inches in the directional pedals.

Cyclic Control System

3. The cyclic control system (fig. 3) consists of dual-tandem cyclic sticks attached to individual support assemblies in each cockpit. The support assembly houses the primary longitudinal and lateral control stops, and two linear variable differential transformers (LVDTs) designed to measure electrically the longitudinal and lateral motions of the cyclic for DASE computer inputs. A series of push-pull tubes and bellcranks transmit the motion of the cyclic control to the servoactuators and the mixer assembly. Motion of the mixer assembly positions the nonrotating swashplate, which transmits the control inputs to the rotating swashplate to control the main rotor blades in cyclic pitch (fig. 4). The cyclic stick grips are shown in figure 5. A stick fold linkage is provided to allow the copilot/gunner (CPG) to lower the cyclic stick to prevent interference when operating the weapon systems.

Collective Control System

4. The collective pitch control system (fig. 6) consists of dual-tandem collective sticks which transmit collective inputs to the main rotor through a series of push-pull tubes and bellcranks attached to the collective servoactuator. Motion of

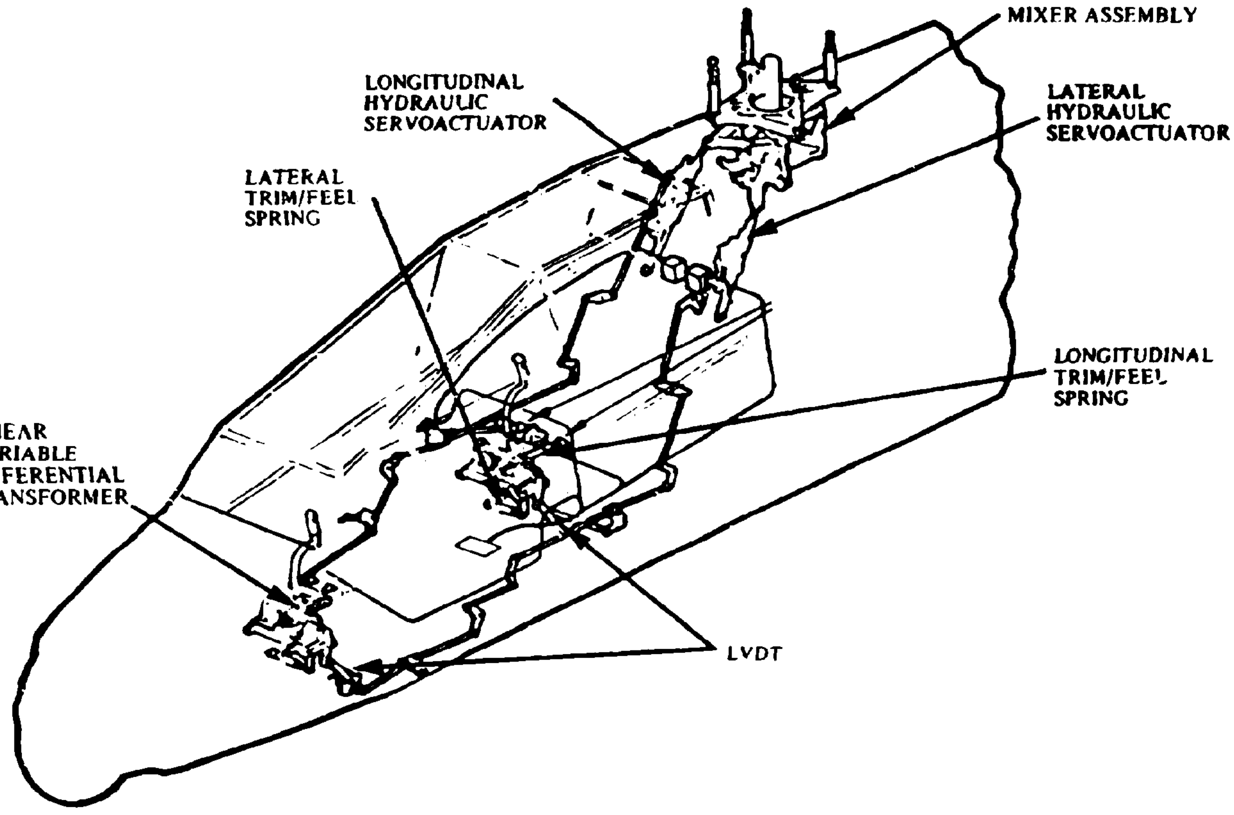


Figure 3. Cyclic Control System

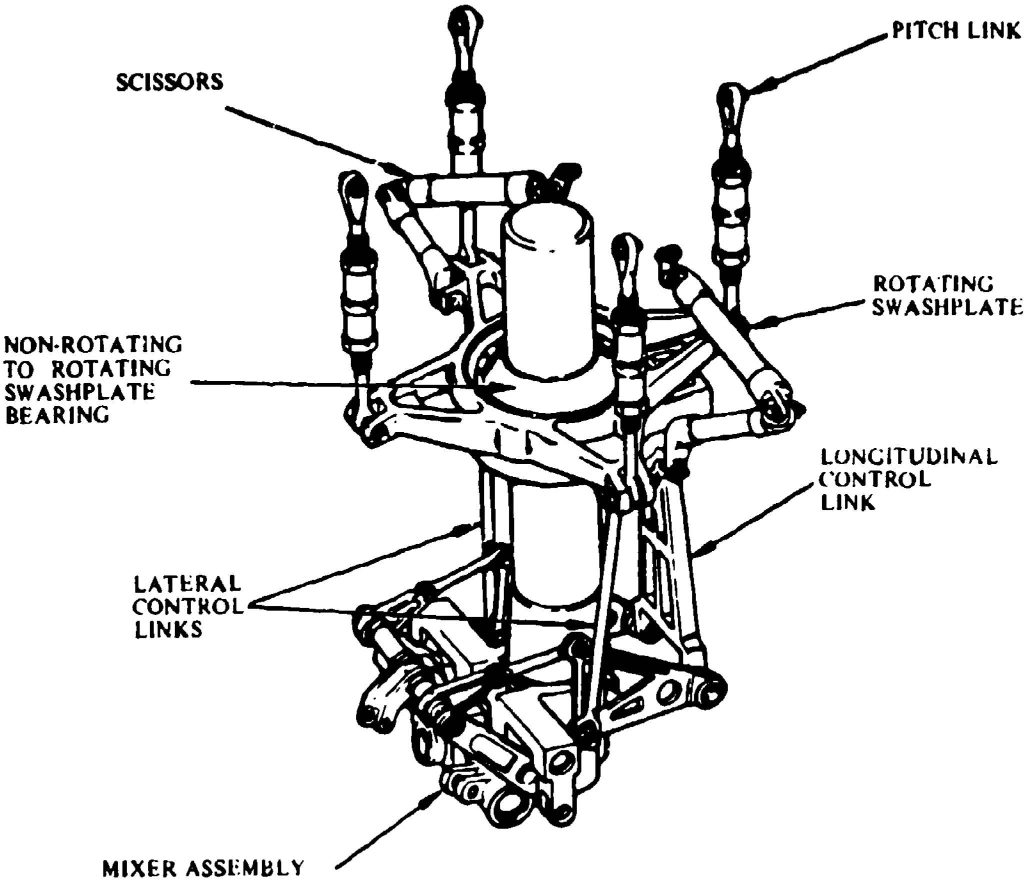
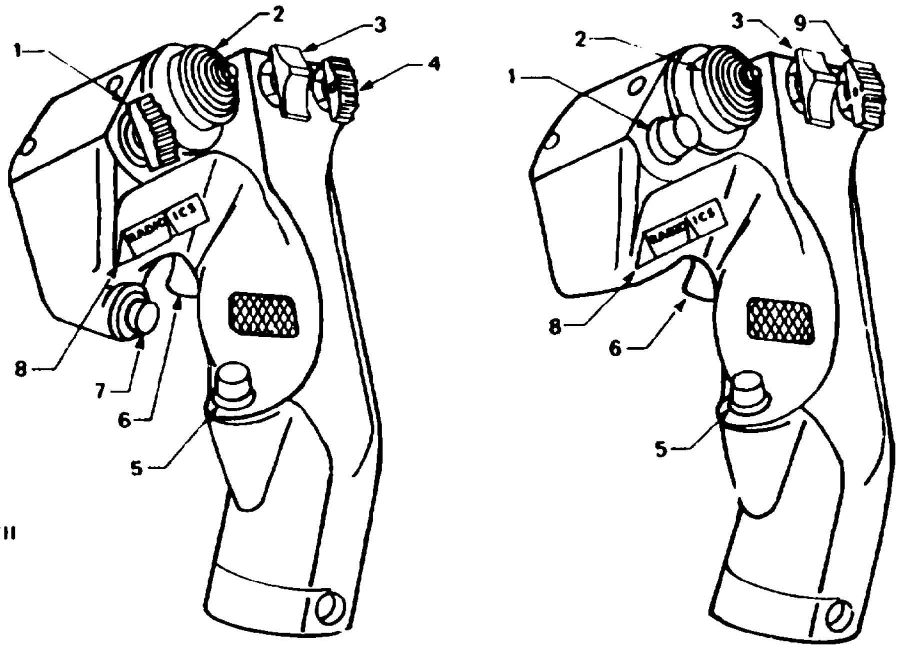


Figure 4. Main Rotor Swashplate Assembly

PILOT CYCLIC STICK GRIP

COPILOT/GUNNER CYCLIC STICK GRIP



1 TRIM RELEASE SWITCH (MOMENTARY ONLY)

2 WEAPONS ACTION SWITCH

3 FLIGHT MODE SYMBOLOGY SWITCH

4 TO BE DETERMINED

5 DASE RELEASE SWITCH

6 GUARDED TRIGGER SWITCH

7 REMOTE TRANSMITTER SELECTOR SWITCH (PILOT GRIP ONLY)

8 RADIO, ICS ROCKER SWITCH

9 NEAR/FAR FOCUS

Figure 5. Cyclic Stick Grips

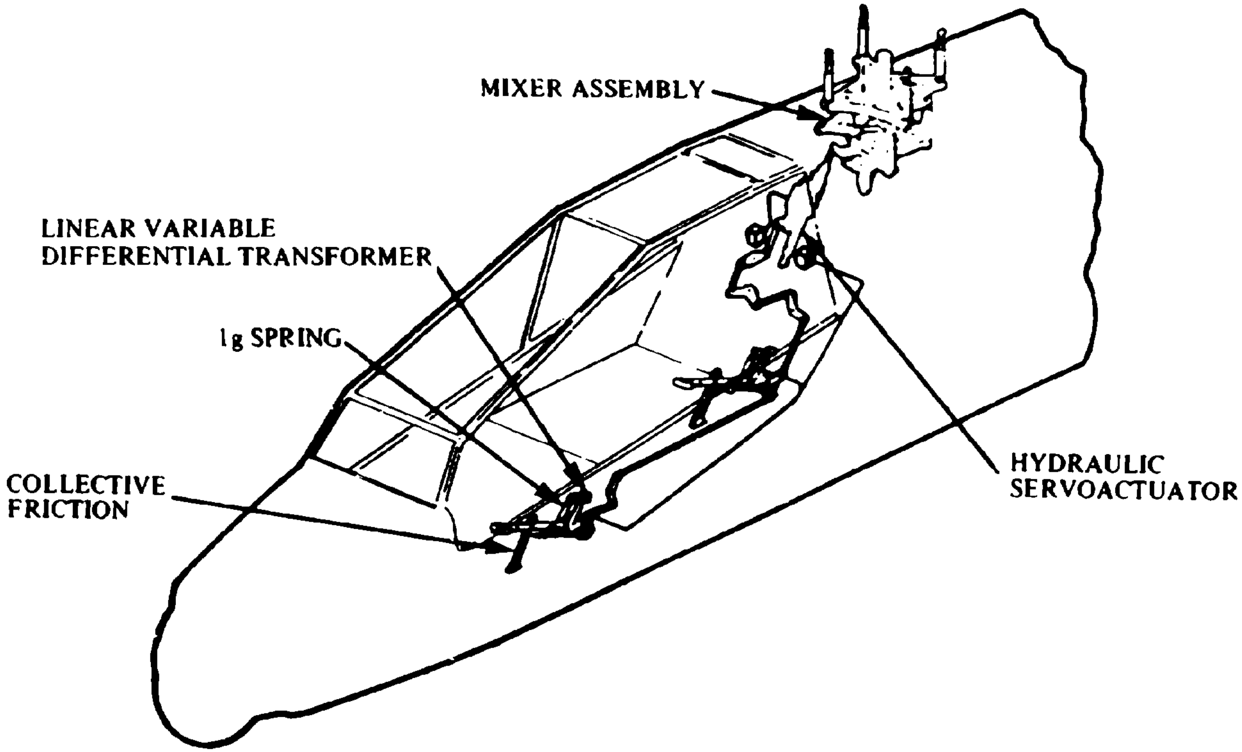


Figure 6. Collective Control Subsystem

the servoactuator is transmitted through the mixer assembly to the swashplate to control the main rotor blades in collective pitch. Collective inputs are also transmitted to the load demand spindle of each engine hydromechanical unit (HMU). The HMU meters the fuel as appropriate to provide collective pitch compensation. Located at each collective control base assembly are the primary control stop, an LVDT, and a one g balance spring. The LVDT supplies electrical inputs to the stabilator control units.

5. Each collective stick (fig. 7) incorporates a switch box assembly, an engine chop collar, a stabilator control panel and an adjustable friction control. The engine chop collar allows rapid deceleration of both engines to flight idle, primarily to allow immediate action in the event of a tail rotor failure.

Directional Control System

6. The directional control system (fig. 8) consists of a series of push-pull tubes and bellcranks which transmit directional pedal inputs to the tail rotor hydraulic servoactuator located in the vertical stabilizer. Attached to each directional pedal assembly is a primary tail rotor control stop and one LVDT. Two sets of wheel brake cylinders are attached to the directional pedals and a 360 degree swiveling tail wheel is incorporated. The tail wheel may be locked in the trailing position by means of a switch located on the pilot instrument panel.

Trim Feel System

7. A trim feel system is incorporated in the longitudinal, lateral, and directional control systems. The system uses individual magnetic brake clutch assemblies in each of the control linkages. Trim feel springs are incorporated to provide a control force gradient and positive control centering. The electromagnetic brake clutch is powered by 28 VDC and is protected by the TRIM circuit breaker. A complete DC electrical failure will disable the trim feel system and allow the cyclic and directional pedals to move freely without resistance from the trim feel springs. The trim release switch on the pilot and CPG grip allows momentary release of the trim feel system.

Horizontal Stabilator

8. The horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. A dual, series, 28 VDC electromechanical actuator allows incidence changes of +45 to -10 degrees leading edge up (LEU) of travel. Safety features include an automatic shutdown capability which allows operation in the

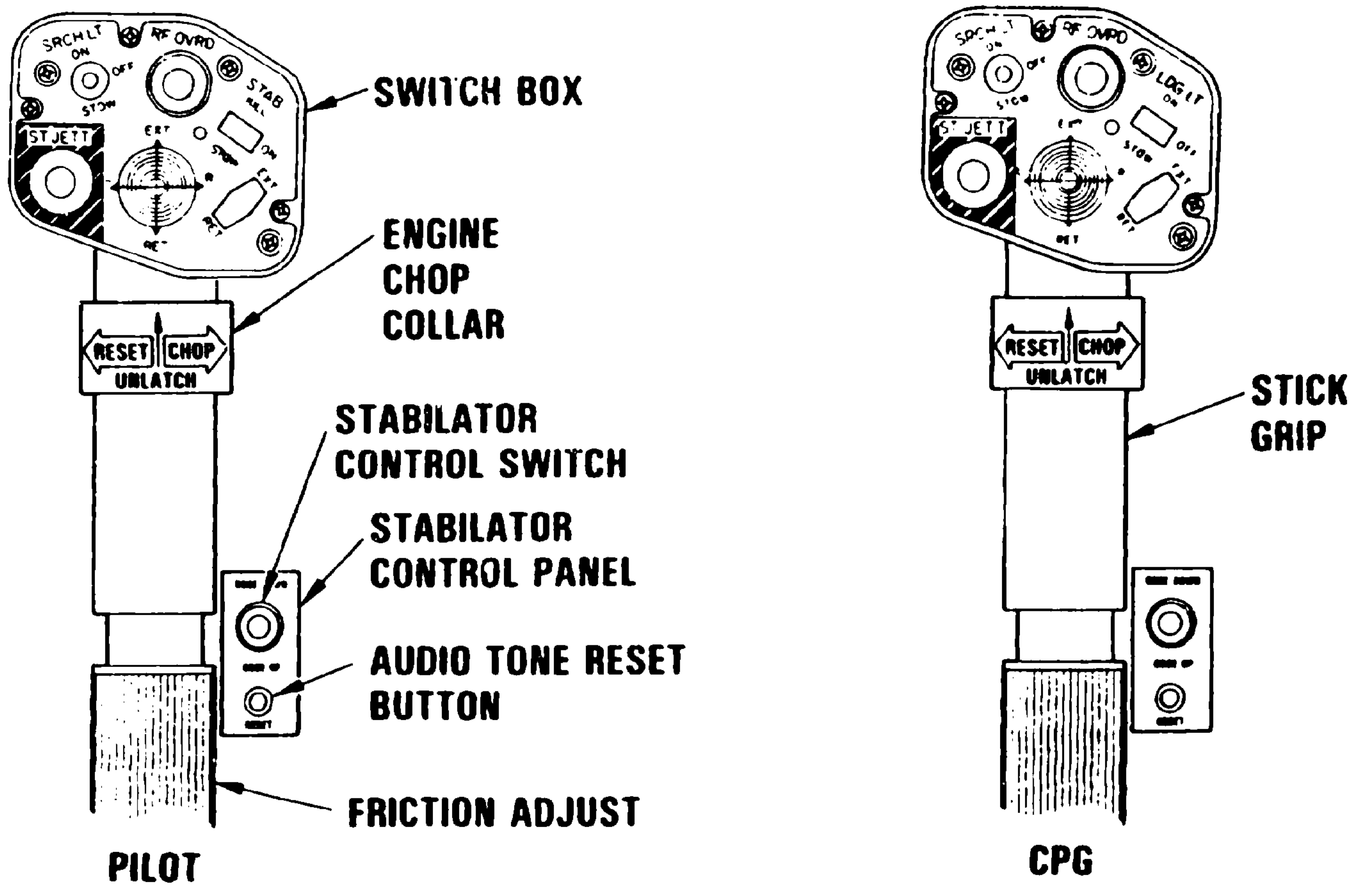


Figure 7. Collective Stick and Switch Box

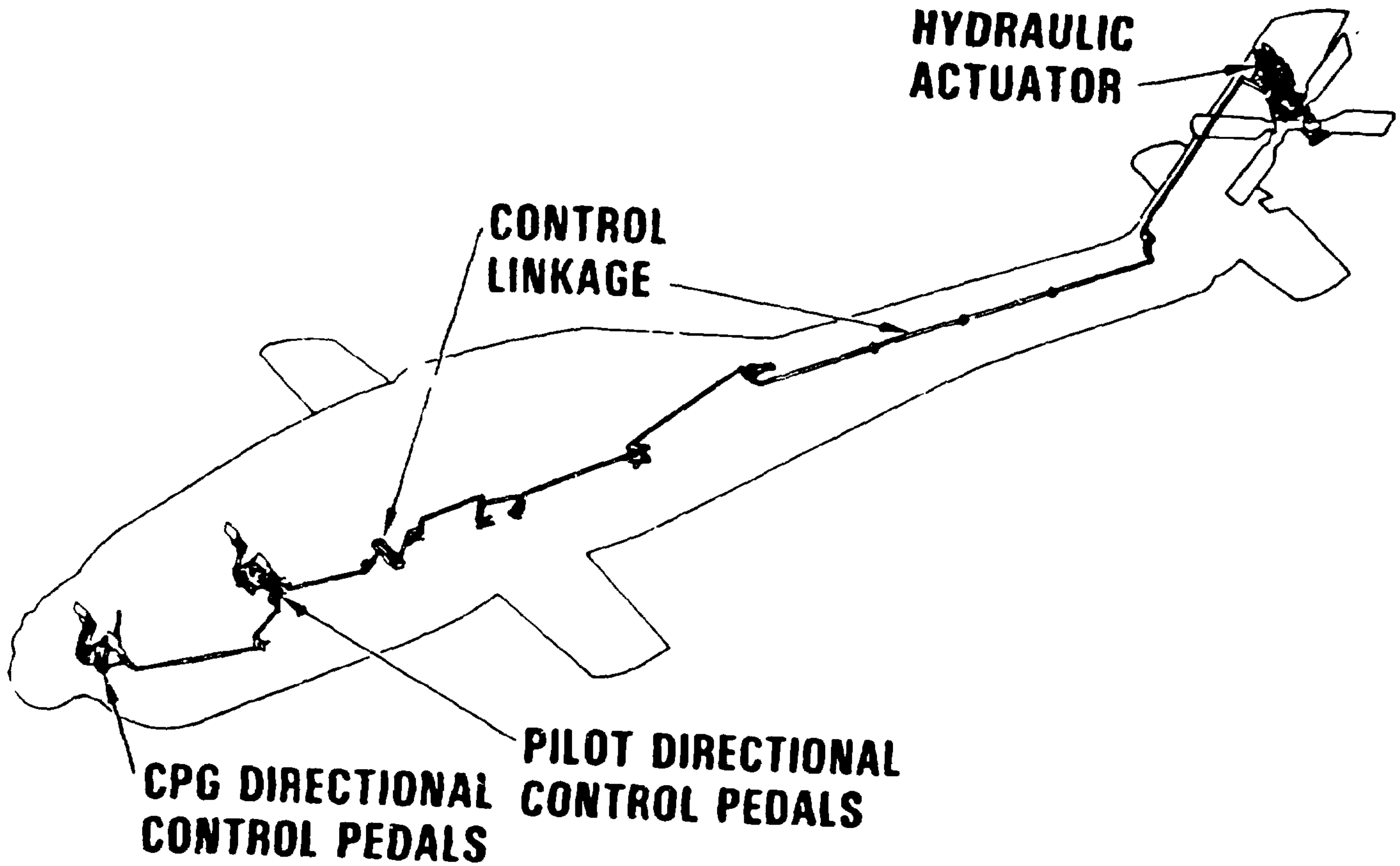


Figure 8. Directional Controls

manual mode by means of a stabilator control panel located on each collective stick. An audio tone is associated with the failure of the automatic mode of operation. A stabilator kill switch, located on the pilot collective stick, disables both the automatic and manual modes operation to protect against a hardover failure (USA S/N 77-23258) only. There are three modes of stabilator operation: the automatic mode, the Nap-of-the-earth (NOE)/ Approach mode and the manual mode. The stabilator is controlled in the automatic mode by two stabilator control units (SCUs). Each SCU controls one side of the dual actuator. Both SCUs receive collective control position information from redundant LVDTs. Two independent pitch rate gyros provide pitch rate information to the SCUs (one gyro for each SCU). The Air Data System (ADS) provides airspeed to both SCUs. Additionally, the left-hand pitot-static system supplies airspeed to one SCU and the right-hand system provides airspeed to the other SCU. Both SCUs receive position information from both sides of the dual actuator. The maximum rate of stabilator travel is 7 degrees per second (deg/sec).

9. The automatic mode is operational when the aircraft has normal AC and DC electrical power applied. Automatic positioning of the stabilator during flight is primarily a function of airspeed and collective position. The stabilator also responds with a low gain (0.2 deg/sec/sec) and limited authority ($\pm$5.0 deg) to pitch rate inputs to the SCU. Software in the SCU limits the incidence change in the automatic mode from +25 deg to -5 deg LEU.

10. The NOE/Approach Mode is selected through the NOE/APPR mode switch on the pilot DASE panel and will stay engaged at any speed. The mode becomes operational below 80 knots indicated airspeed (KIAS) and will move the stabilator to 25 degrees LEU at a 3.6 deg/sec rate. The mode can be disengaged by manual mode selection below 80 knots, activation of the DASE release, or by the AUTO STAB reset switch. Acceleration through 80 KIAS will engage the automatic schedule and the stabilator will move at a 3.6 deg/sec rate for 10 seconds or until commanded position is reached. The stabilator then reverts to normal automatic schedule of 7 deg/sec. Failure to revert to automatic schedule will result in system disengagement with both visual and aural indications.

11. The manual mode can be selected below 80 KIAS through the pilot and CPG manual control switch on either collective stick. Manual control selection will result in STAB FAIL caution light illumination. Selection of automatic mode can be accomplished by pressing the AUTO STAB reset switch on the pilot or CPG collective stick. The stabilator will move at a 3.6 deg/sec

rate for the first 10 seconds or until the automatic mode schedule position is reached. Acceleration through 80 KIAS in manual mode will engage the normal automatic mode and the stabilator will move at a 7 deg/sec rate.

12. The SCUs have a fault detection feature which will switch the stabilator mode of operation from automatic to manual if any of the following conditions are sensed:

a. A mismatch between the positions of the two sides of the actuator equivalent to 10 degrees of stabilator travel (if there is a runway failure of one side of the actuator, this feature will disable the automatic mode after 10 degrees of stabilator movement).

b. The stabilator at a position of 20 degrees or greater with an airspeed greater than 110 KIAS.

c. The stabilator at a position of 30 degrees or greater with an airspeed greater than 80 KIAS (for 1 second or longer).

d. AC power to the collective position LVDTs less than 23 VAC.

Flight Control Rigging

13. A flight control rigging check was performed in accordance with procedures outlined in HHI Experimental Test Procedure (ETP) 7-211510000, dated 1 August 79 (main rotor upper controls), ETP 7-211520000, dated 3 March 81 (tail rotor powered controls), and ETP 7-211123600, dated 21 April 80 (stabilator control rigging) in conjunction with Stabilator Checkout Procedure (rev J4) dated 30 July 82. The horizontal stabilator schedule is shown in figure 9. Tables 2 and 3 present the collective and cyclic rigging. Tail rotor rigging is shown below:

Full right pedal: 15.1 degrees thrust to left
Full left pedal: 27.0 degrees thrust to right

Digital Automatic Stabilization Equipment

14. The DASE provides rate damping (SAS), command augmentation (CAS), HAS, attitude hold, and turn coordination. A block diagram of the DASE is provided as figure 10. The DASE is controlled by the digital automatic stabilization equipment computer (DASEC). The DASEC receives information from several sources. The heading and attitude reference system (HARS) provides the DASEC with aircraft angular velocities (3 axes), aircraft attitudes (pitch, roll, and heading), and inertial horizontal and lateral velocities

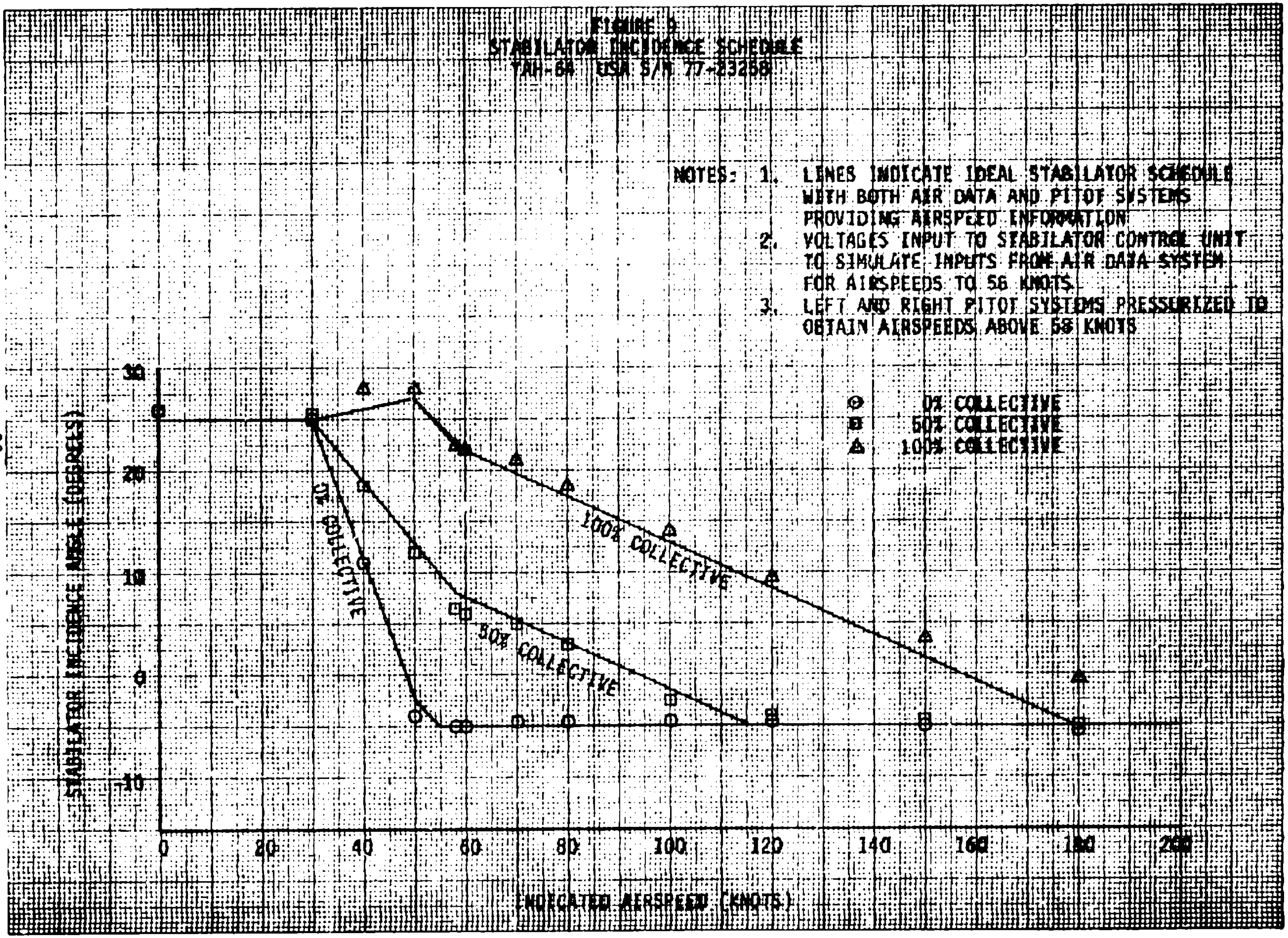


FIGURE 9
STABILATOR INCIDENCE SCHEDULE
YAH-64 USA S/N 77-23258
NOTES: 1. LINES INDICATE IDEAL STABILATOR SCHEDULE WITH BOTH AIR DATA AND PITOT SYSTEMS PROVIDING AIRSPEED INFORMATION
2. VOLTAGES INPUT TO STABILATOR CONTROL UNIT TO SIMULATE INPUTS FROM AIR DATA SYSTEM FOR AIRSPEEDS TO 56 KNOTS
3. LEFT AND RIGHT PITOT SYSTEMS PRESSURIZED TO OBTAIN AIRSPEEDS ABOVE 58 KNOTS
0% COLLECTIVE
50% COLLECTIVE
100% COLLECTIVE
100% COLLECTIVE
50% COLLECTIVE
0% COLLECTIVE
STABILATOR INCIDENCE ANGLE (DEGREES)
30
20
10
0
-10
0
20
40
60
80
100
120
140
160
180
200
INDICATED AIRSPEED (KNOTS)

Table 2. Angle Measurements
Pilots Collective and Cyclic Controls

Blade Azimuth Position (deg)	Item	Rig Pins			Stick position			Measured Clinometer Angle (deg)	Leading Edge Up or Down
		Collective	Longitudinal Cyclic	Lateral Cyclic	Collective	Longitudinal	Lateral		
$\psi = 90$	1	In	In	In	Rig	Rig	Rig	0.8	Down
	2	In	Out	In	Rig	Fwd	Rig	20.0	Down
	3	In	Out	In	Rig	Aft	Rig	11.4	Up
	4	In	In	In	Rig	Rig	Rig	-1.1	Down
	5	Out	In	In	Up	Rig	Rig	10.8	Up
	6	Out	In	In	Down	Rig	Rig	8.4	Down
	7	In	In	In	Rig	Rig	Rig	-1.0	Down
$\psi = 270$	8	In	In	In	Rig	Rig	Rig	-0.7	Down
	9	In	Out	In	Rig	Fwd	Rig	21.5	Up
	10	In	Out	In	Rig	Aft	Rig	10.3	Down
	11	In	In	In	Rig	Rig	Rig	-0.1	Up
$\psi = 0$	12	In	In	In	Rig	Rig	Rig	-1.0	Down
	13	In	In	Out	Rig	Rig	Left	11.8	Up
	14	In	In	Out	Rig	Rig	Right	6.8	Down
	15	In	In	In	Rig	Rig	Rig	-0.7	Down
$\psi = 180$	16	In	In	In	Rig	Rig	Rig	-0.8	Down
	17	In	In	Out	Rig	Rig	Left	10.6	Up
	18	In	In	Out	Rig	Rig	Right	8.5	Down
	19	In	In	In	Rig	Rig	Rig	-1.2	

Table 3. Computation of Blade Angle Travel Pilots Collective and Cyclic Controls

Computation	Travel (deg)	Tolerance (deg)
LONGITUDINAL CYCLIC		
1. Forward = 1/2 (Item *9 - Item 2)= (If Item 2 is leading edge down add item 2)	20.8	20° (minimum)
2. Aft = 1/2 (Item 3 - Item 10) = (If Item 10 is leading edge down add Item 10)	10.9	10° (minimum)
LATERAL CYCLIC		
3. Left = 1/2 (Item 13 - Item 17) = (If Item 17 is leading edge down add Item 17)	11.2	10.5° (minimum)
4. Right = 1/2 (Item 18 - Item 14) = (If Item 14 is leading edge down add Item 14)	7.6	7.0° (minimum)
COLLECTIVE		
5. Full pitch travel = (Item 5-Item 6)= (If Item 6 is leading edge down add Item 6)	19.2	18.0° (minimum)
6. Measured @ pitch housing (Bolt pad machined surface 2.4 in. inboard of lead-lag hinge)	-8.4	-10° to - 7°

*Item numbers obtained from table 2

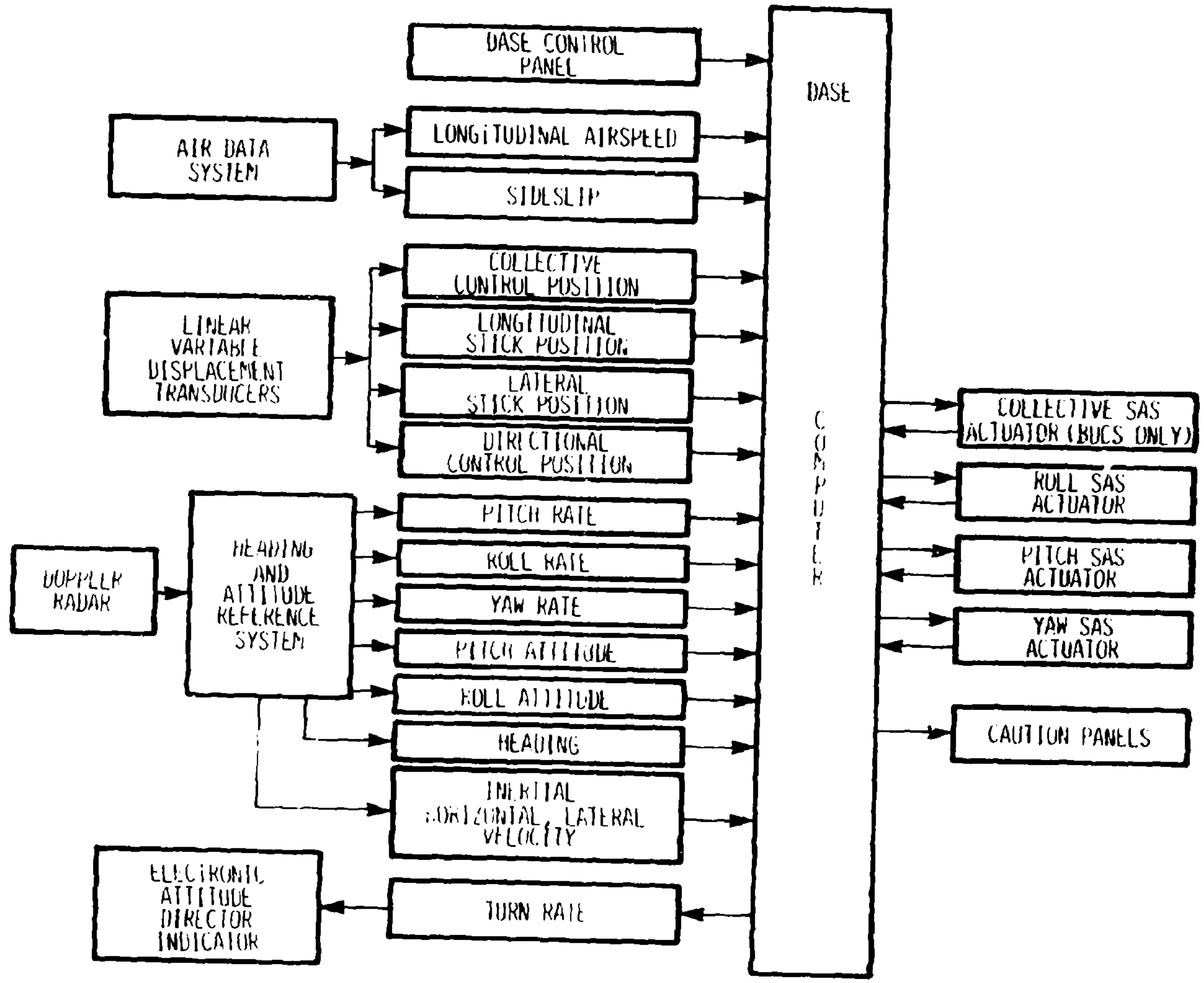


Figure 10. Digital Automatic Stabilization Equipment Block Diagram

(measured by the Doppler radar). The ADS provides longitudinal airspeed and sideslip angle. The LVDTs provide longitudinal, lateral, collective and directional control position information. The DASEC processes this information and commands control inputs through the electrohydraulic servo valves on the longitudinal, lateral, and directional servoactuators. The DASE authority is limited in the lateral and directional axis to ± 10 percent of full control authority while the longitudinal axis is limited to 20 percent forward and 10 percent aft.

15. The SAS function of the DASE system provides rate damping in pitch, roll, and yaw axes. Each axis is separately engageable through a magnetically held toggle switch on the DASE control panel shown in figure 11. The CAS is used to augment the pilot control inputs and is an automatic function of the DASE whenever pitch and roll SAS are selected and yaw SAS is selected below 60 knots true airspeed (KTAS). The yaw CAS function is automatically disengaged during ground operations. Schematic diagrams showing gains and transfer functions of SAS/CAS are provided as figures 12 through 14.

16. A limited authority HAS mode is provided through pitch and roll DASE channels using rates, attitudes and doppler corrected inertial velocities from the HARS. HAS is used to reduce pilot workload by assisting the pilot in maintaining a desired hover position. HAS is engageable below 15 knots ground speed and 50 KTAS whenever the pitch or roll DASE channels are engaged. Additionally, a heading hold mode is provided through the yaw DASE channel using aircraft heading information from the HARS. This function is engaged whenever the yaw DASE channel is engaged and the HAS switch is selected. Schematic diagrams are shown in figures 15 through 17.

17. A limited authority attitude hold mode is provided through pitch and roll SAS. Attitude Hold is engageable above 60 KTAS whenever pitch and roll SAS are engaged. Attitude Hold will automatically disengage whenever the airspeed is decreased to 50 KTAS. Schematic diagrams are shown in figures 18 and 19.

18. A limited authority turn coordination function is provided through yaw SAS using sideslip information from the ADS. This function is automatically provided above 60 KTAS whenever yaw SAS is engaged. A schematic diagram is shown in figure 20.

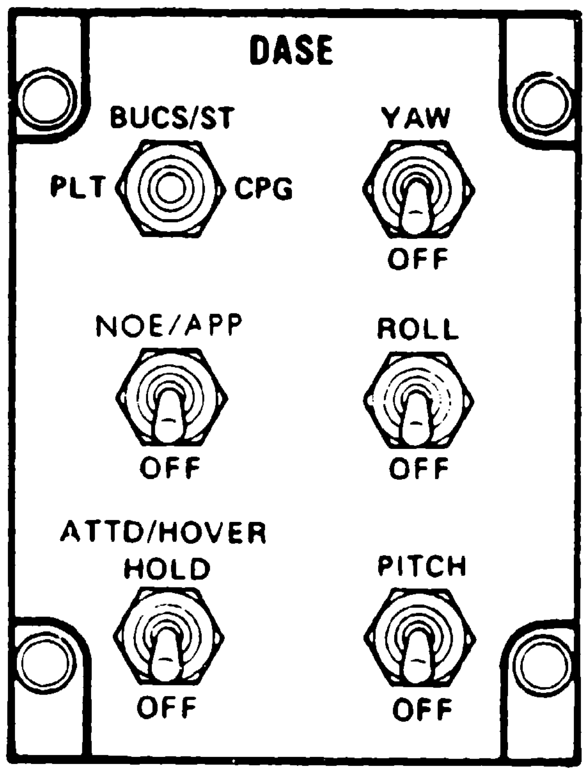
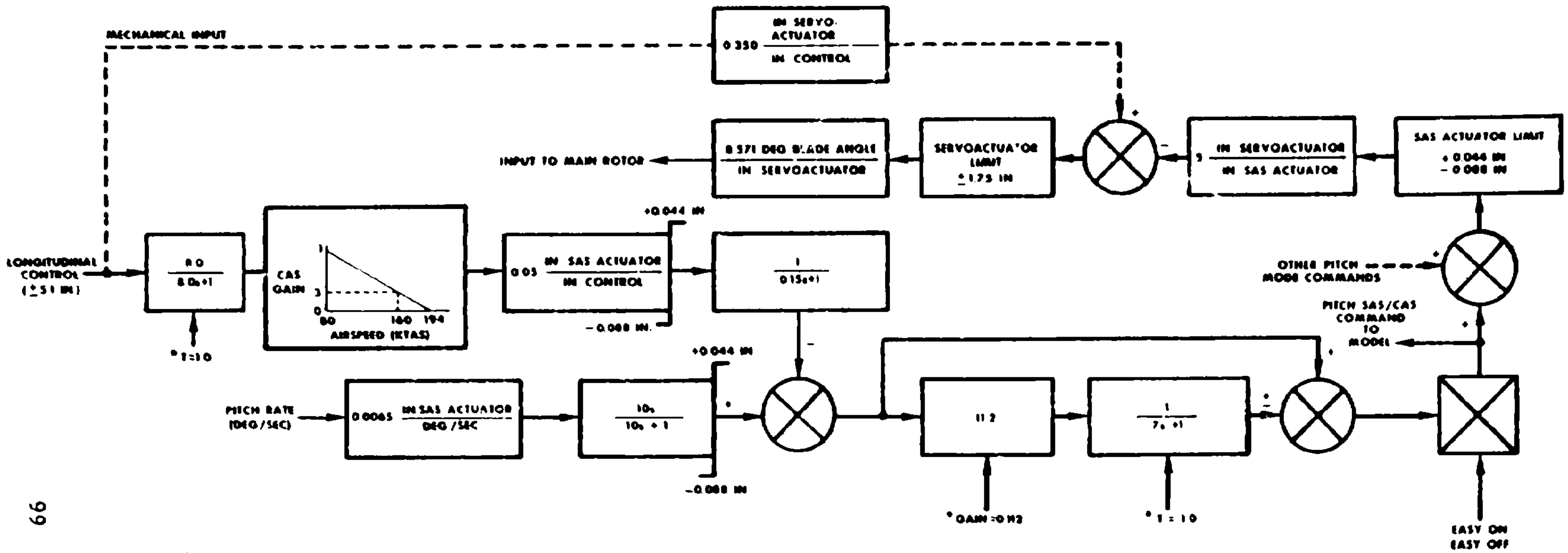


Figure 11. DASE Control Panel



* RESYNCHRONIZED MODE ONLY, ACTIVATED BY TRIM RELEASE SWITCH (τ = TIME CONSTANT)

FIGURE 12. PITCH SAS/CAS MODE

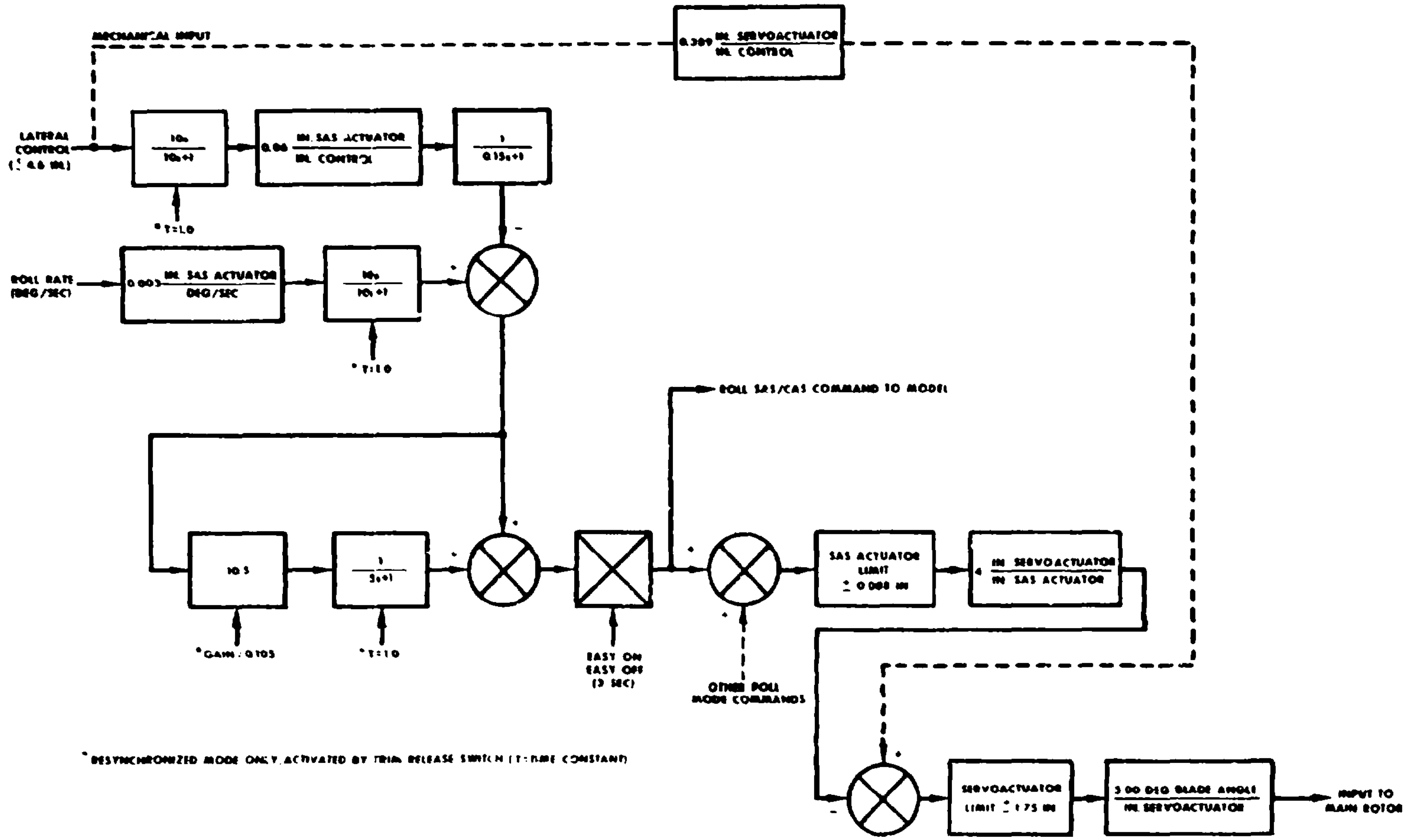


FIGURE 13. ROLL SAS/CAS MODE

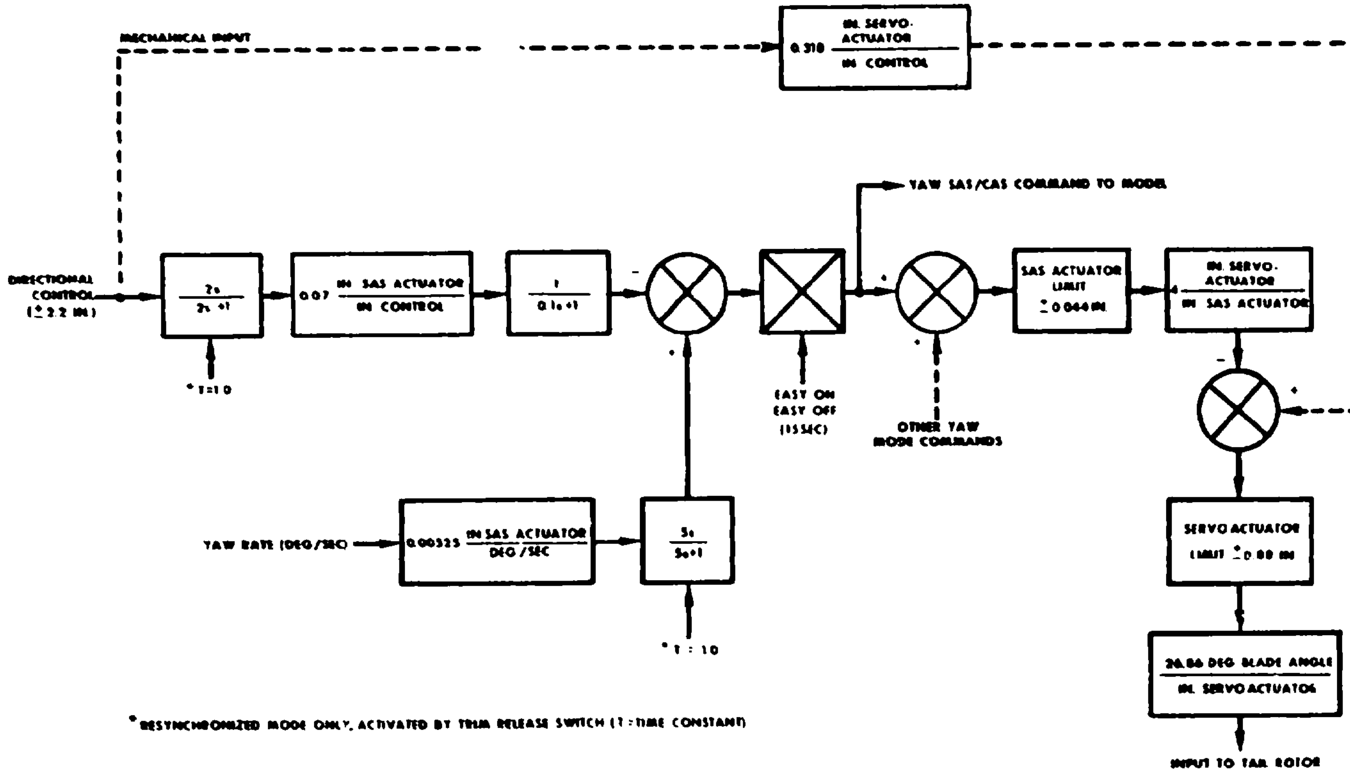


FIGURE 14. YAW SAS/CAS LOW SPEED MODE

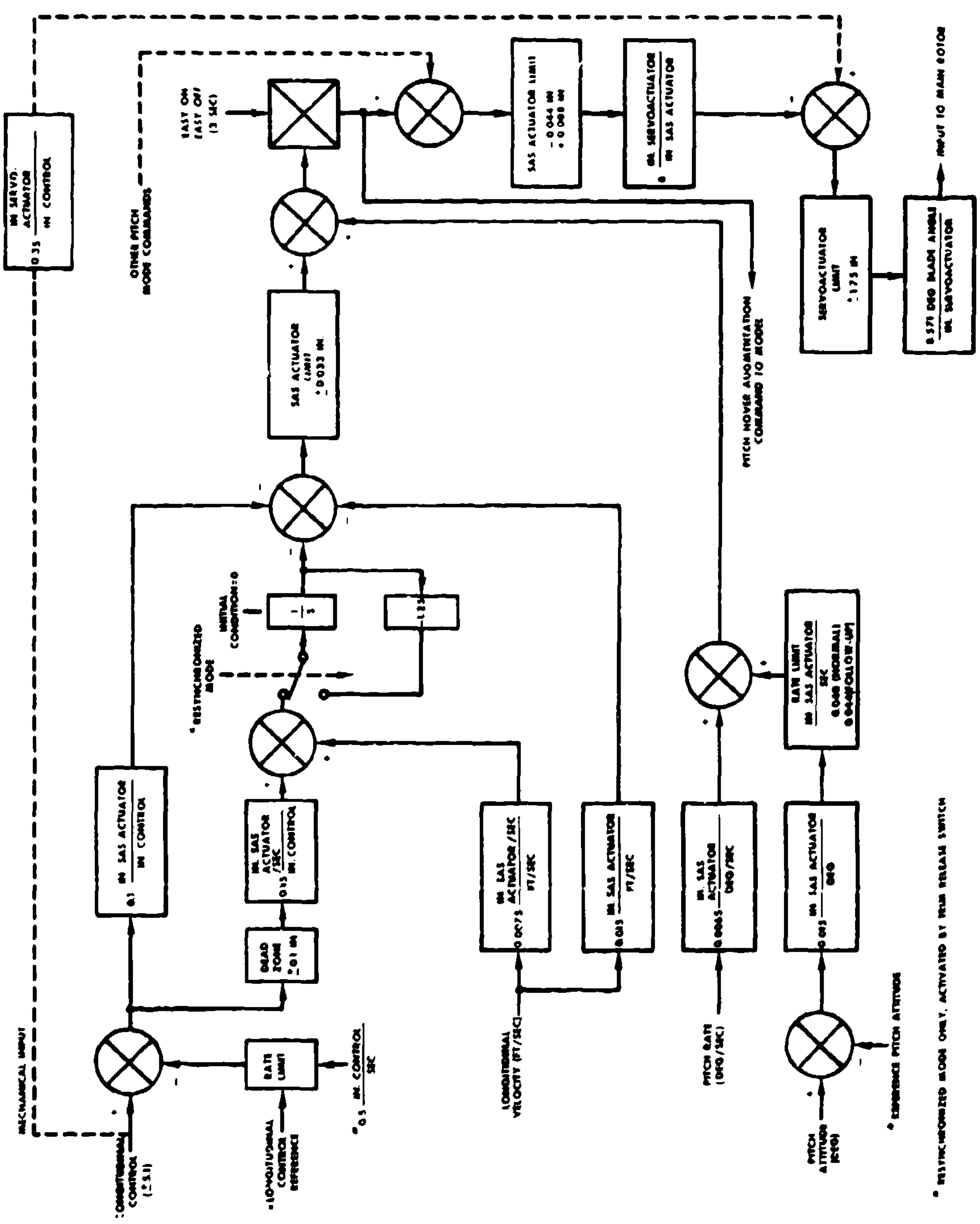
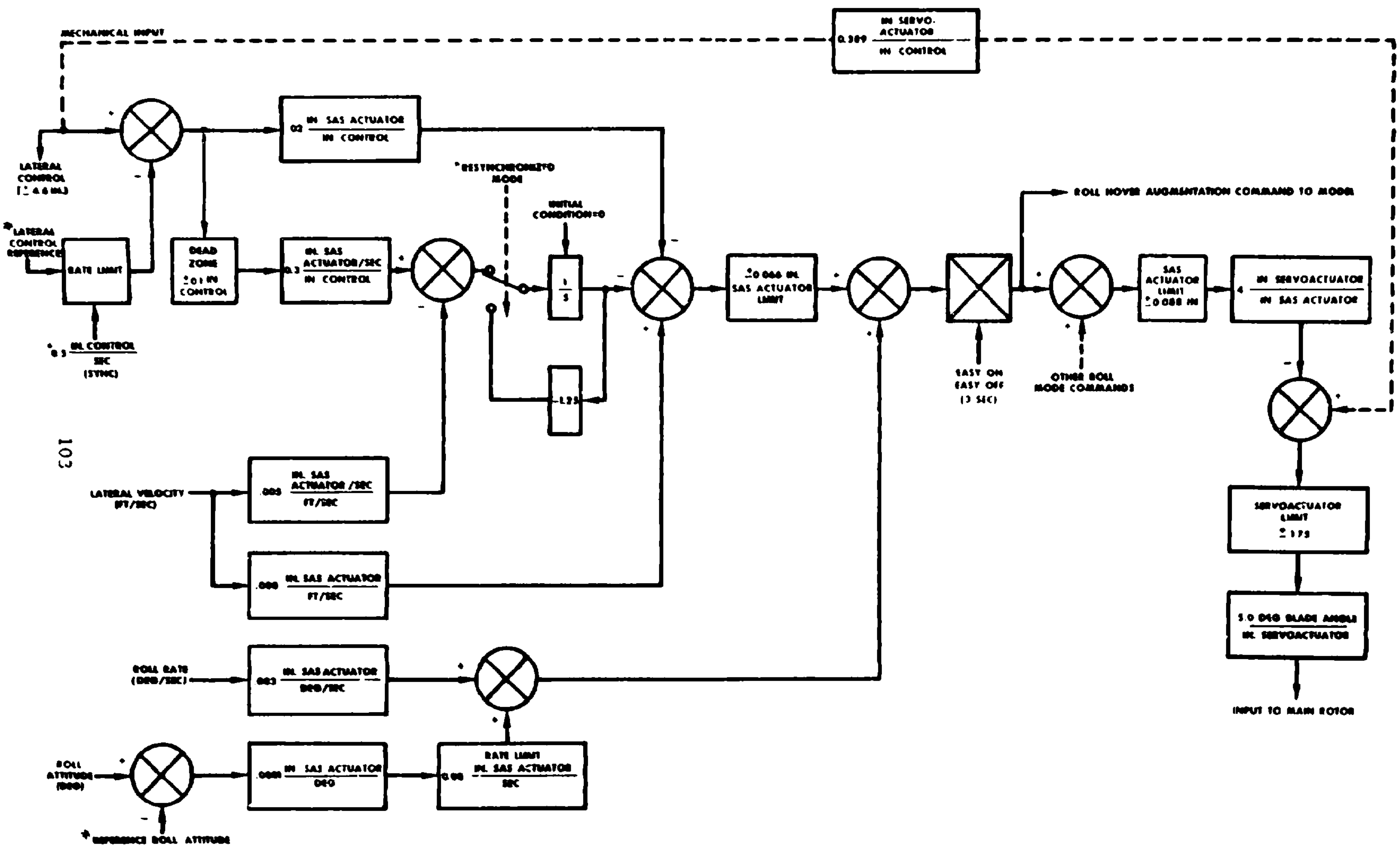


FIGURE 15. PITCH HOVER AUGMENTATION MODE



* RESYNCHRONIZED MODE ONLY. ACTIVATED BY TRIM RELEASE SWITCH

FIGURE 16. ROLL HOVER AUGMENTATION MODE

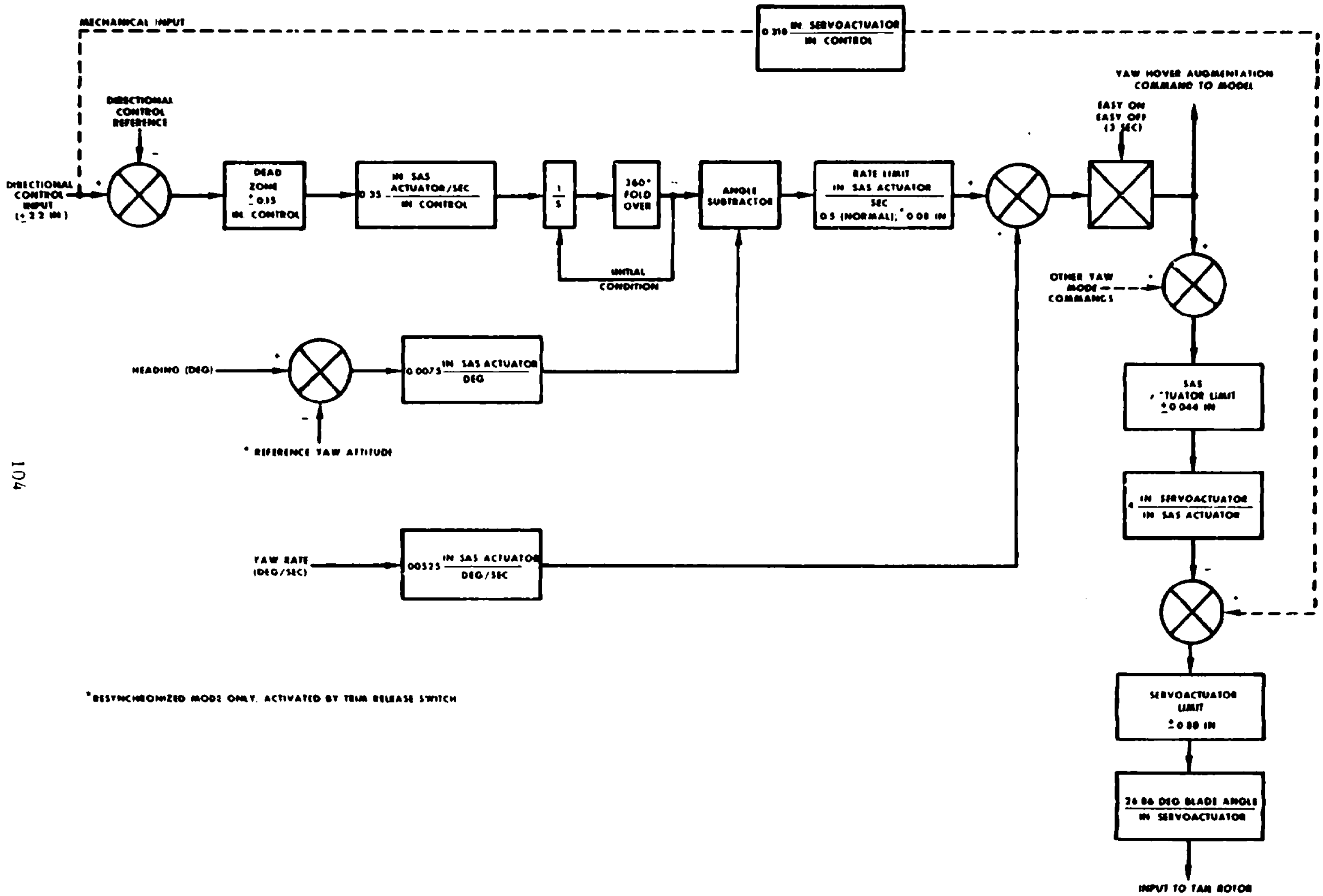
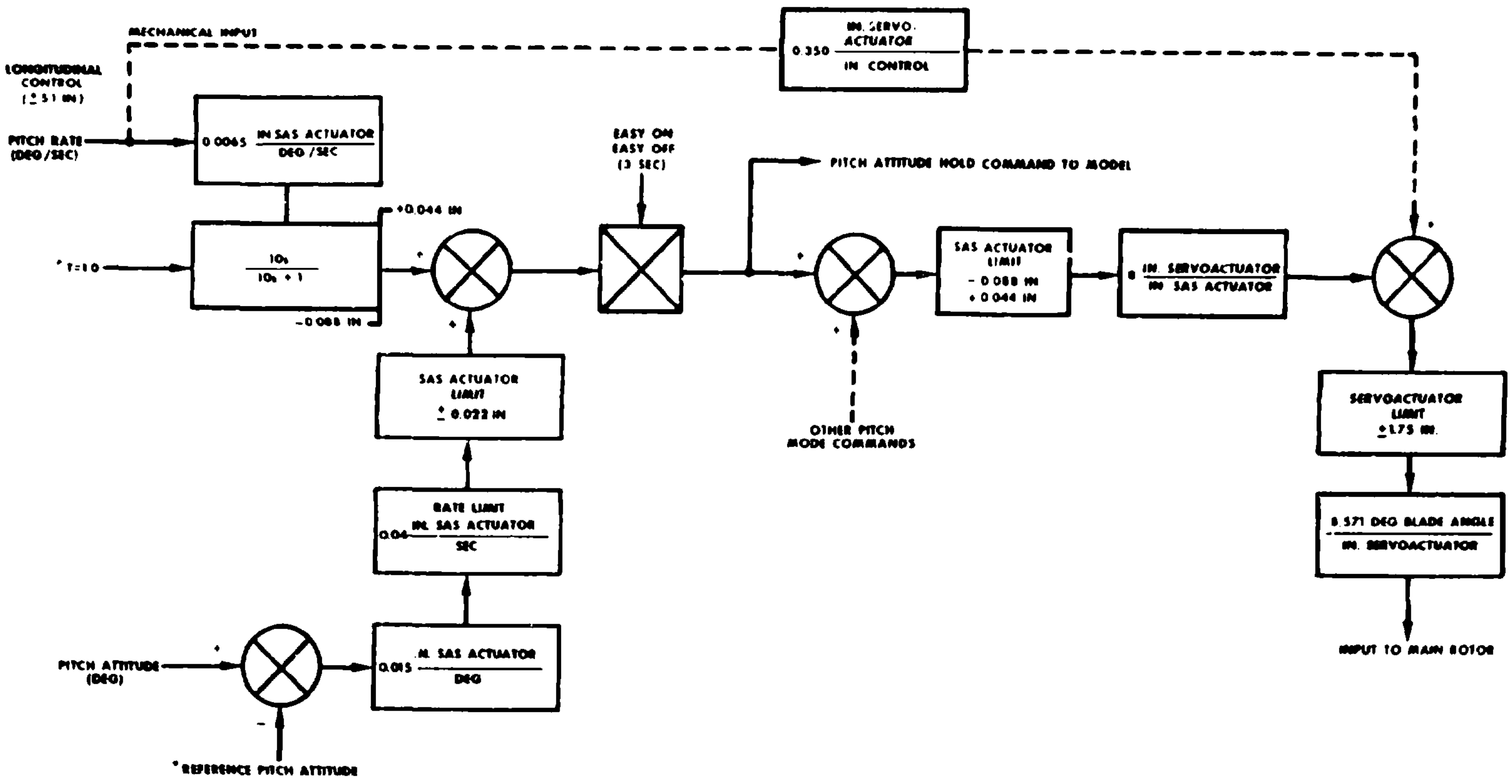
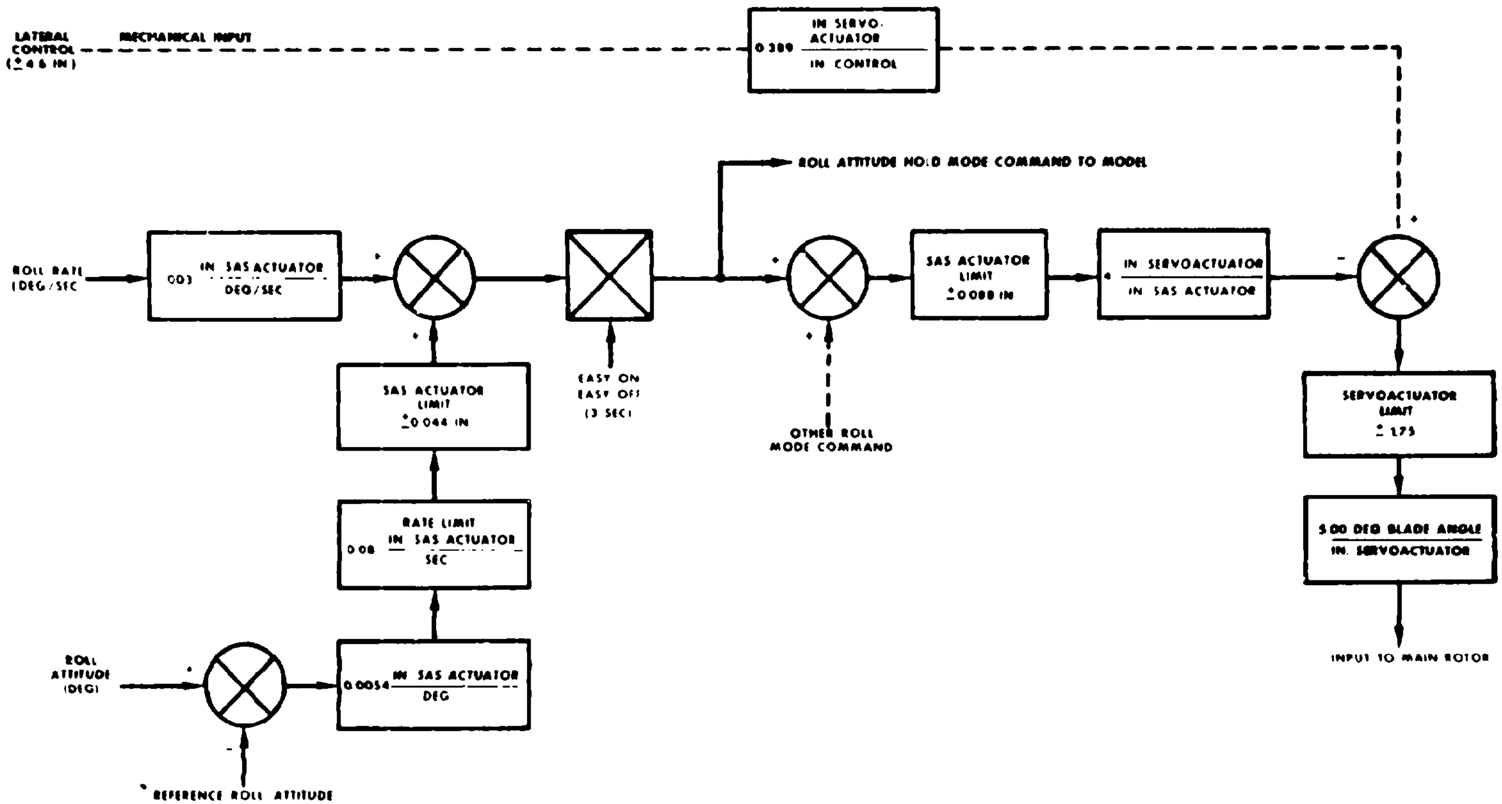


FIGURE 17. HEADING HOLD MODE



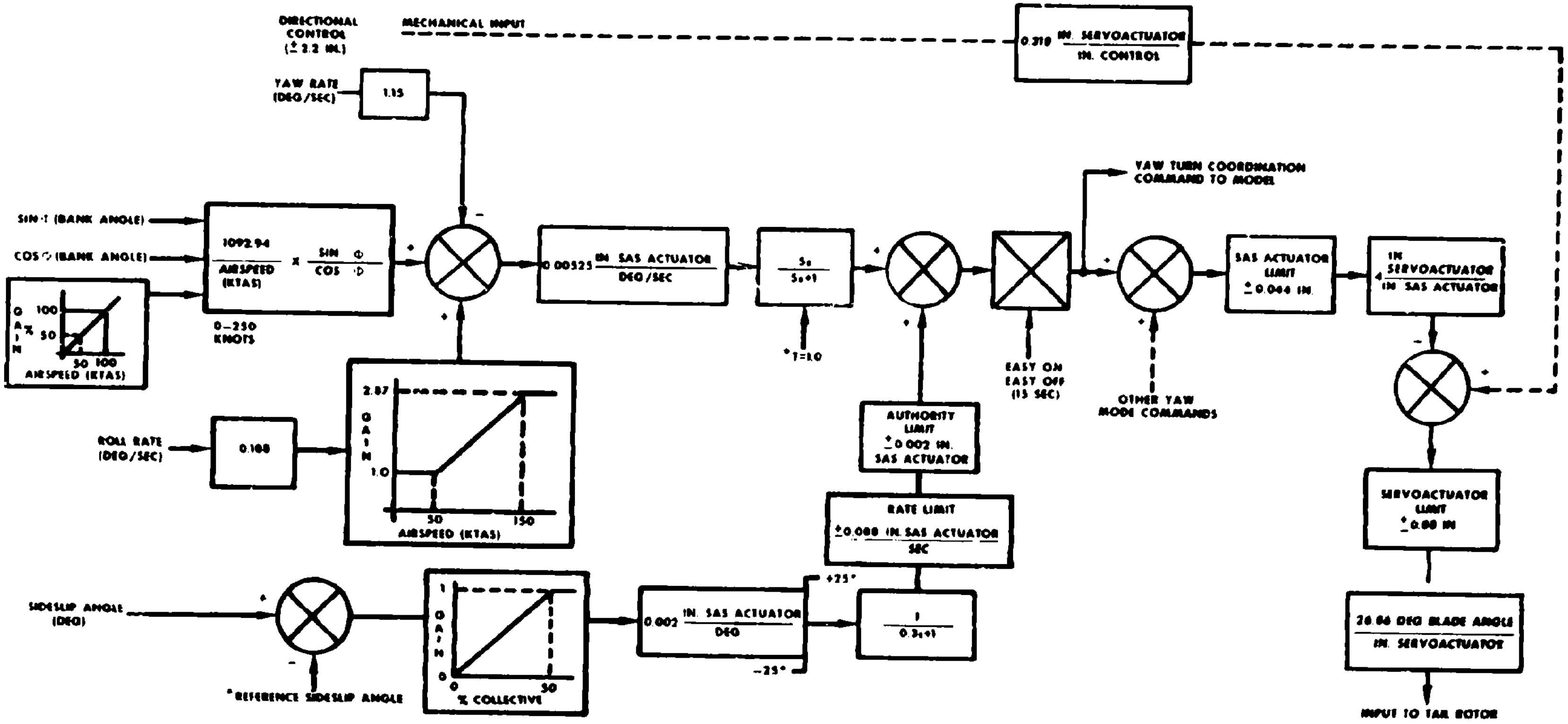
* RESYNCHRONIZED MODE ONLY, ACTIVATED BY TIME RELEASE SWITCH (T=TIME CONSTANT)

FIGURE 18. PITCH ATTITUDE HOLD MODE



* RESYNCHRONIZED MODE ONLY. ACTIVATED BY TRIM RELEASE SWITCH

FIGURE 19. ROLL ATTITUDE HOLD MODE



*RESYNCHRONIZED MODE ONLY, ACTIVATED BY TRIM RELEASE SWITCH (T=TIME CONSTANT)

Φ =BANK ANGLE

FIGURE 20. YAW TURN COORDINATION MODE

HYDRAULIC SYSTEM

General

19. The hydraulic system consists of four hydraulic servoactuators powered simultaneously by two independent 3000-psi hydraulic systems. Each servoactuator simultaneously receives pressure from the primary and utility systems to drive the dual-tandem actuators. This design allows the remaining system to automatically continue powering the servos in the event of a single hydraulic system failure. The two systems (primary and utility) are driven by the accessory gearbox utilizing variable displacement pumps, independent reservoirs and accumulators. The APU drives all accessories, including the hydraulic pumps, when the aircraft is on the ground and the rotors are not turning. The accessory gearbox is driven by the main transmission during flight and provides for normal operation of both hydraulic systems during autorotation. An emergency hydraulic system is provided to allow emergency operation of the flight controls in the event of a dual system failure.

Primary Hydraulic System

20. The primary hydraulic system (fig. 21) consists of a one-pint capacity reservoir, which is pressurized to 30 psi using air from the shaft-driven compressor; an accumulator, which has a nitrogen precharge of 1600 psi, designed to reduce surges in the hydraulic system; and a primary manifold that directs the fluid to the lower side of the four serovactuators. The primary system also provides the hydraulic pressure for operation of the DASE functions.

Utility Hydraulic System

21. The utility hydraulic system (fig. 22) consists of an air pressurized 1.3 gallon reservoir and a 3000-psi accumulator which drives the APU starting motor. The utility manifold directs fluid to the upper side of the servoactuators, the stores pylon system, tail wheel lock mechanism, area weapon turret drive, and rotor brake. Other manifold functions include an auxiliary isolation check valve which isolates the area weapon turret drive and external stores actuators when either a low pressure or low fluid condition exists: a low pressure sensor isolates the accumulator as an emergency hydraulic source for the servoactuators in the event of a dual hydraulic system failure. The accumulator assembly stores enough fluid for emergency operation of the flight controls through four full strokes of the collective stick and one 180 degrees heading change. The emergency system may be activated by either the pilot or CPG emergency switch. An electrically

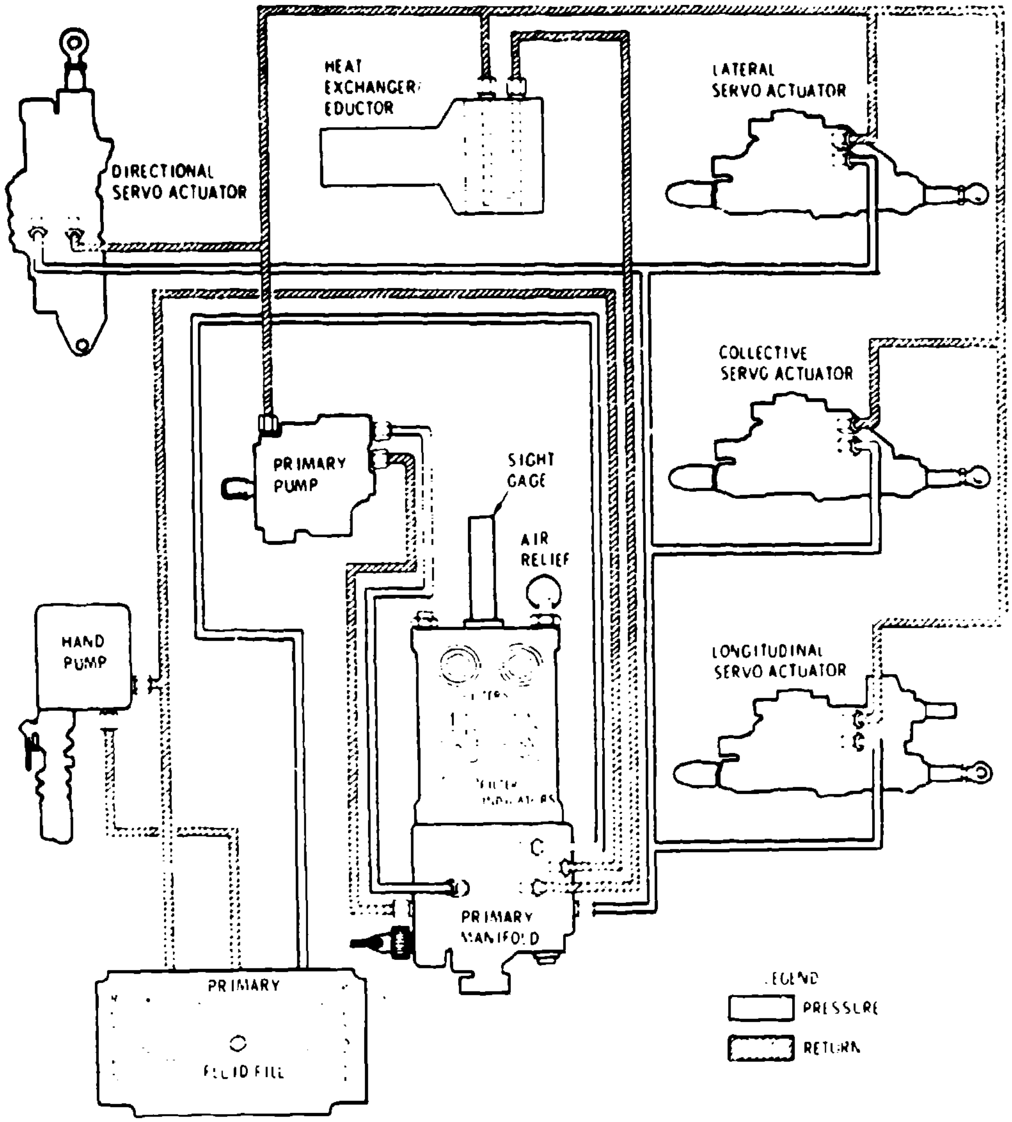


Figure 21. Primary Hydraulic System

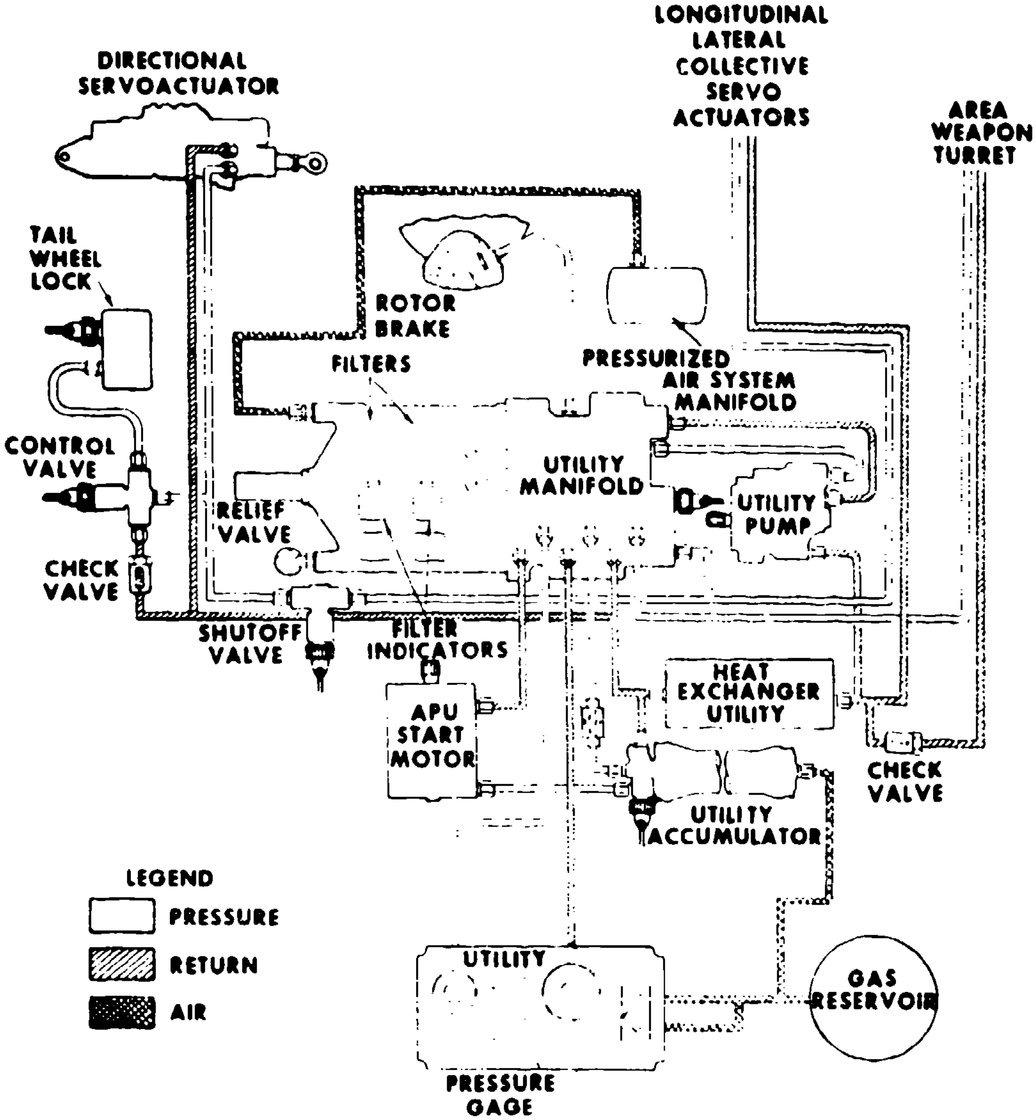


Figure 22. Utility Hydraulic System

activated emergency shutoff valve is designed to isolate the utility side of the directional servoactuator and the tail wheel lock mechanism when a low fluid condition exists. The electrical connections for this valve were not installed on S/N 77-23258 during this test.

Servoactuators

22. Individual hydraulic servoactuators are provided for longitudinal, lateral, collective, and directional controls. Each servoactuator (fig. 23) consists of a ballistically tolerant housing, a single actuator rod and dual frangible pistons, a BUCS plunger, and various parts for routing of both primary and utility hydraulic fluid. The system is designed to accomodate all flight loads with a failure of either system, however, some control authority will be lost in the directional servoactuator system. DASE and BUCS functions would be lost with failure of the primary system. The BUCS plunger assemblies were installed during this test, however, electrical connections were omitted.

ENGINES

23. The YAH-64 helicopter, for this test, was powered by two General Electric YT700-GE-701 front drive turboshaft engines, rated at 1690 shp (sea level, standard day, uninstalled). The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: a cold section, a hot section, a power turbine, and an accessory section. Design features of each engine include an axial-centrifugal flow compressor, a through-flow combustor, a two-stage air-cooled high-pressure gas generator turbine, a two-stage uncooled power turbine and self-contained lubrication and electrical systems. In order to reduce sand and dust erosion, and foreign object damage, an integral particle separator operates when the engine is running. The YT700-GE-701 engine also incorporates a history recorder which records total engine events. Engines S/N GE-E-374002 and GE-E-374001 were installed in the left and right positions, respectively. Both engines were equipped with G02J Electrical Control Units (ECUs). ECUs S/N 0008 and 2004 were installed on the left and right engine, respectively. The following engine data are provided:

Model	YT700-GE-701
Type	Turboshaft

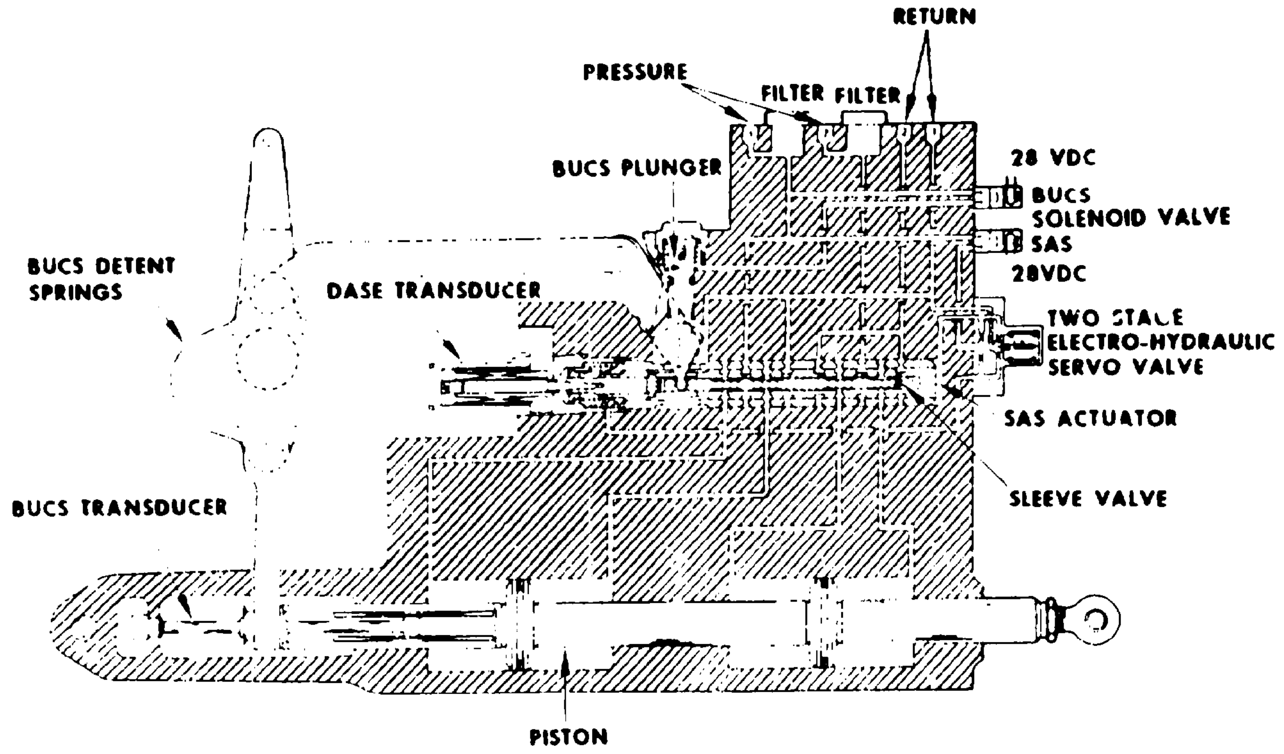


Figure 23. Flight Control Servoactuator

Rated power (intermediate)	1690 shp sea level, standard day, uninstalled
Output speed (at 100 percent N_R)	20,952 RPM
Compressor	5 axial stages, 1 centrifugal stage
Variable geometry	Inlet guide vanes, stages 1 and 2 stator vanes
Combustion chamber	Single annular chamber with axial flow
Gas generator turbine stages	2
Power turbine stages	2
Direction of rotation (aft looking forward)	Clockwise
Weight (dry)	423 lb
Length	47 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624 (JP-4 or JP-5)
Lubric[illegible] oil	MIL-L-7808 or MIL-L-23699
Electrical power requirements for history recorded and N_P overspeed protection	40W, 115 VAC, 400 Hz
Electrical power requirements for anti-ice valve, filter bypass indication, oil filter bypass indication, and magnetic chip detector	1 amp, 28 VDC

INFRARED (IR) SUPPRESSION SYSTEM

24. The IR suppression system consists of finned exhaust pipes attached to the engine outlet and bent outboard to mask hot engine parts. The finned pipes radiate heat which is cooled by rotor downwash in hover and turbulent air flow in forward flight. The engine exhaust plume is cooled by mixing it with engine cooling air and bay cooling air (fig. 24). The exhaust acts as an eductor, creating air flow over the combustion section of the engine providing engine cooling. Fixed louvers on the top and bottom of the aft cowl and a door on the bottom forward cowling provide convective cooling to the engine during shutdown. The movable bottom door is closed by engine bleed air during engine operation.

FUEL SYSTEM

25. The YAH-64 fuel system has two fuel cells located fore and aft of the ammunition bay. The system includes a fuel boost pump in the aft cell for starting and for high-altitude operation, a fuel transfer pump for transferring fuel between cells, a fuel crossfeed/shutoff valve, and provisions for pressure and gravity fueling and defueling. Additionally, provisions exist for external, wing-mounted fuel tanks. The two ferry configuration for this test included 181 gal. fuel tanks. Figure 25 is a schematic of the fuel system. Figure 26 shows the locations and capacities of the two internal fuel cells.

26. By using the tank select switch on the fuel control panel (fig. 27), the pilot or CPG can select either or both tanks from which the engines will draw fuel. With the tank select switch in the NRML position, the left (No. 1) engine will draw fuel from the forward fuel cell and the right (No. 2) engine will draw from the aft cell. When FROM FWD is selected on the tank select switch, the two fuel crossfeed/shutoff valves are positioned so that both engines draw fuel from the forward tank. The FROM AFT position allows the engines to draw fuel from the aft tank only. The tank select switch is disabled whenever the boost pump is on. When the boost pump is on, the fuel crossfeed/shutoff valves are positioned to allow only fuel from the aft cell to feed both engines. The air-driven boost pump operates automatically during engine start and may be activated by the switch on the pilot or CPG fuel control panel.

27. The pilot and CPG also have the capability to transfer fuel between tanks using the transfer switch on the fuel control panels. Moving the fuel transfer switch out of the OFF position closes the refuel valve and activates the air-driven pump which transfers fuel in the selected direction.

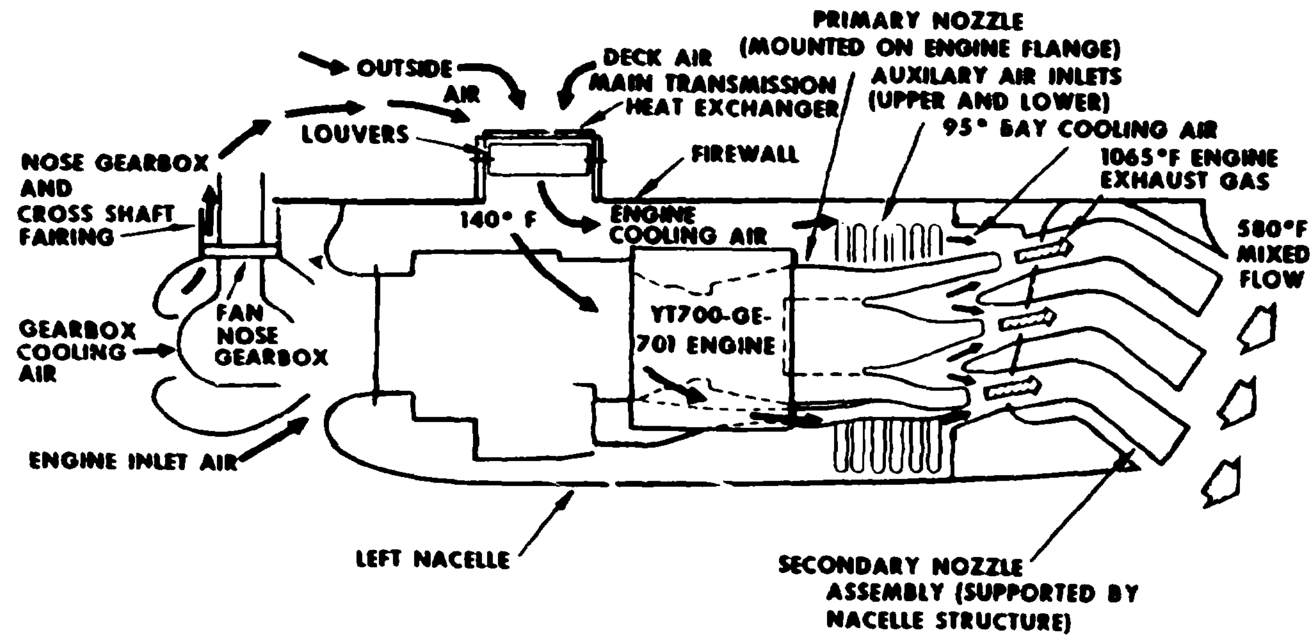


Figure 24. Infrared Suppression System Engine Cooling

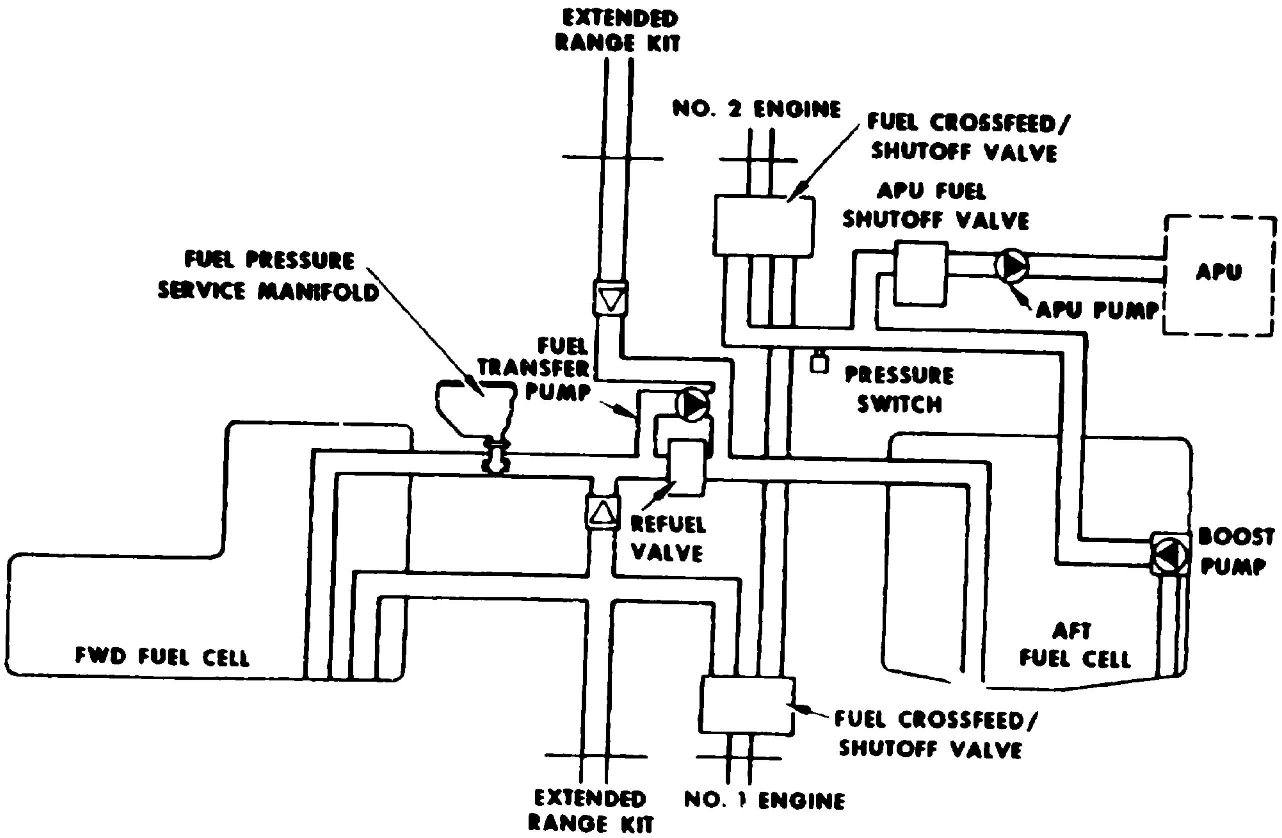


Figure 25. Fuel System Major Components

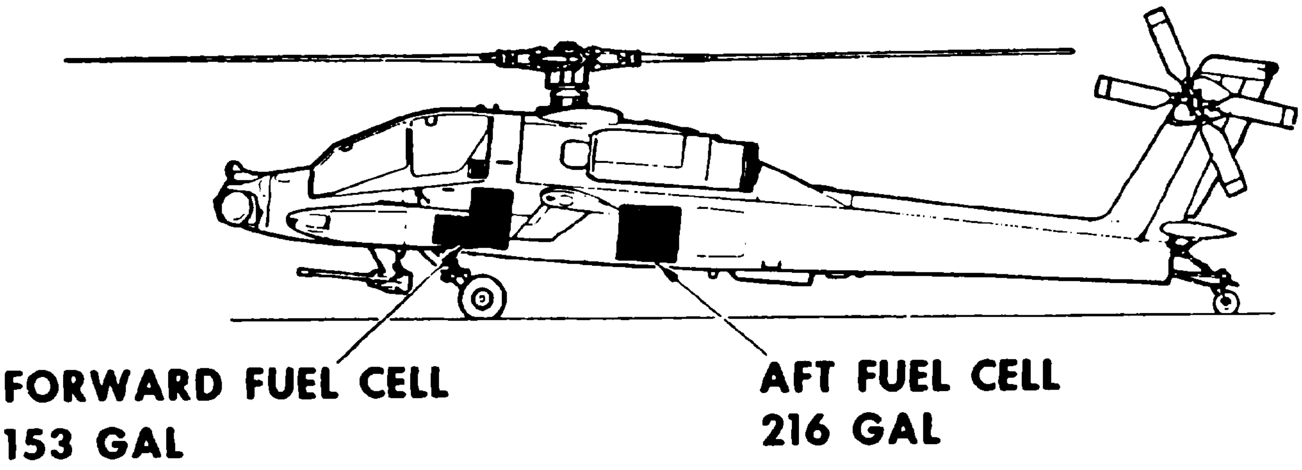


Figure 26. Fuel Cell Locations

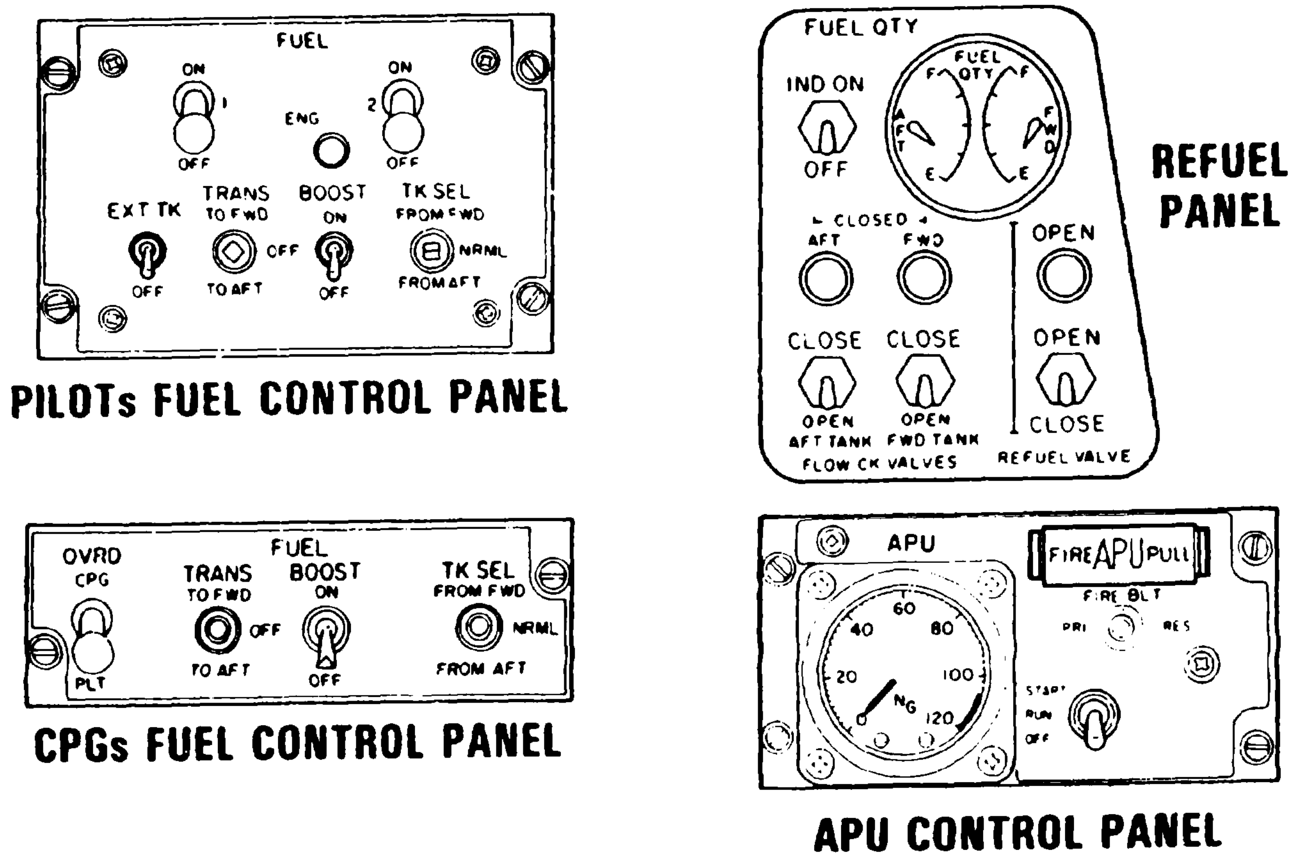


Figure 27. Fuel System Controls

APPENDIX C. INSTRUMENTATION

1. The airborne data acquisition system was installed, calibrated, and maintained by Hughes Helicopters, Inc. The system used pulse code modulation encoding, and magnetic tape was used to record parameters on board the aircraft. A boom was mounted on the left side of the aircraft, extending 52 inches forward of the nose. A swiveling pitot-static tube, an angle-of-attack sensor, and an angle-of-sideslip sensor were mounted on the boom. Figure 1 presents the position error correction for the boom airspeed system. Instrumentation and related special equipment used for this test follows:

<u>Pilot Station (Aft Cockpit)</u>

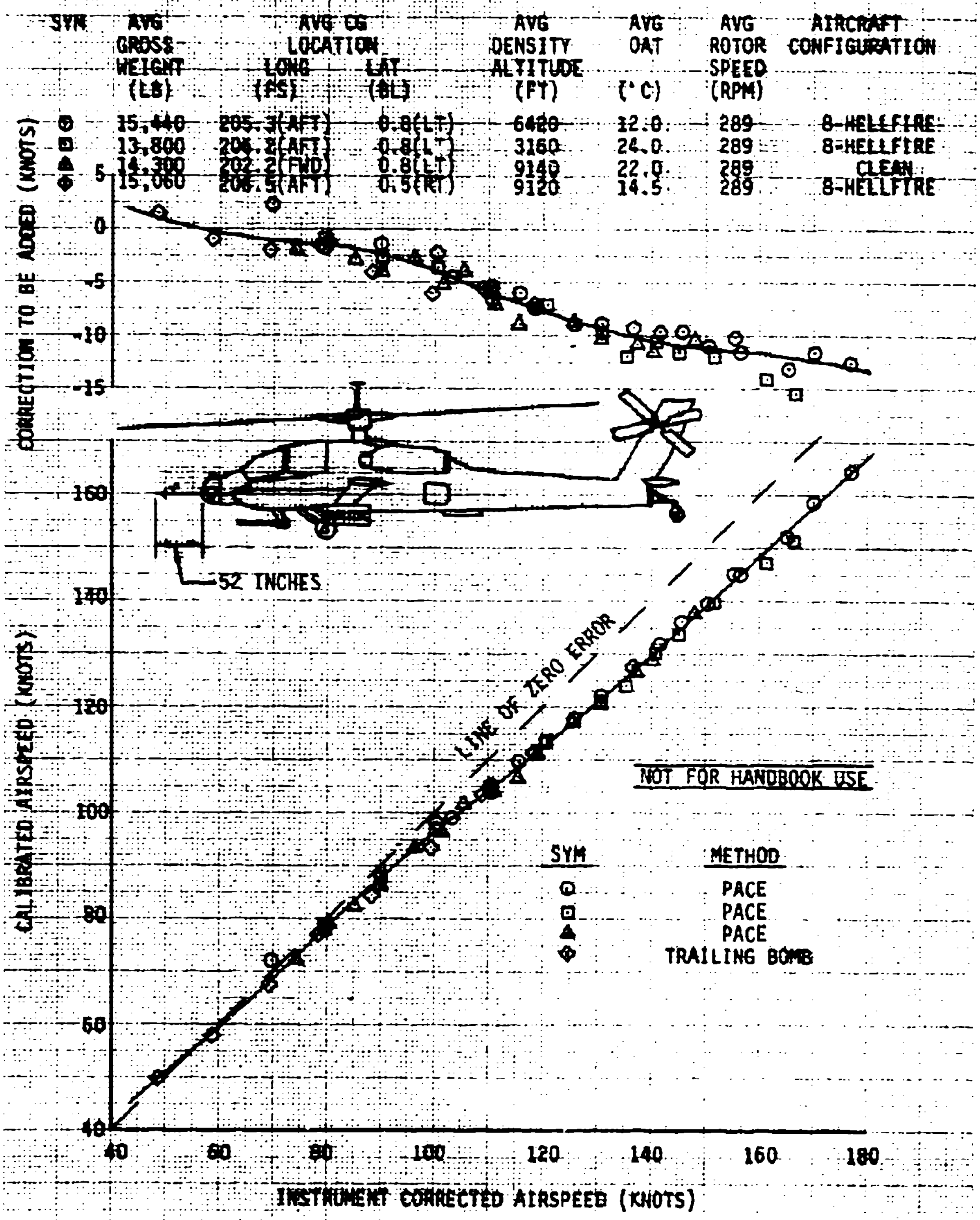
Pressure altitude (boom)
Airspeed (boom) (sensitive and digital)
Tether cable tension
Main rotor speed (digital)
Engine torque (both engines)*
Engine turbine gas temperature (both engines)*
Engine power turbine speed (both engines)*
Engine gas producer speed (both engines)*
Angle of sideslip
Tether cable angles (longitudinal and lateral)
Longitudinal control position
Lateral control position
Directional control position
Collective control position
Stabilator angle*
Normal acceleration

<u>Copilot/Gunner Station (Forward Cockpit)</u>

Airspeed (ship, left)
Altitude (ship)
Main rotor speed
Engine torque (both engines)*
Engine turbine gas temperature (both engines)*
Engine gas producer speed (both engines)*
Total air temperature
Time code display
Data system controls
Fuel ed (both engines)

*Standard ship systems calibrated for test

FIGURE 1
BOOM SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT
YAH-64 USA S/N 77-23258



PCM Parameters

Time code
Event
Main rotor speed
Fuel temperature (both engines)
Fuel used (both engines)
Engine fuel flow rate (both engines)
Engine gas producer speed (both engines)
Engine power turbine speed (both engines)
Engine torque (both engines)
Engine turbine gas temperature (both engines)
Airspeed (boom)
Airspeed (ship, right and left)
Altitude (boom)
Altitude (ship)
Total air temperature (boom)
Angle of attack
Angle of sideslip
Tether cable tension
Tether cable angle (longitudinal and lateral)
Control positions
- Longitudinal cyclic
- Lateral cyclic
- Directional
- Collective

Stabilator incidence angle
Aircraft attitudes (from heading and attitude reference set (HARS))
- Pitch
- Roll
- Yaw

Aircraft angular velocities (from HARS)
- Pitch
- Roll
- Yaw

Stability augmentation system actuator position
- Longitudinal
- Lateral
- Directional

Air data system
- Longitudinal airspeed
- Lateral airspeed
- Resultant airspeed
- Pressure altitude
- Air temperature

Control actuator positions
- Tail rotor
- Collective pitch
- Longitudinal cyclic
- Lateral cyclic

Radar altitude
Center of gravity normal acceleration
Center of gravity lateral acceleration
Vibration accelerometers
- Pilot station (3 axes)(pilot seat)
- Pilot floor (3 axes)
- Copilot station (3 axes) (copilot seat)
- Copilot floor (3 axes)
- Center of gravity (3 axes)

Other Instrumentation

Main rotor 1/rev signal transmitted over VHF radio (noise survey only)

2. A portable weather station, consisting of an anemometer, sensitive temperature gauge, and barometer, was used to record wind speed, wind direction, ambient temperature, and pressure altitude at selected heights up to 100 feet above ground level.

3. A calibrated load cell was incorporated with a cargo hook installed in the aircraft ammunition bay. Indicators were installed in the cockpit for displaying the cable tension and cable angle during the tethered hover tests.

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Performance data were obtained using the methods described in Army Materiel Command Pamphlet AMCP 706-204 (ref 13, app A) as a guide. Handling Qualities Data were evaluated using test methods described in Naval Air Test Center Flight Test Manual FTM No. 101 (ref 14) as a guide.

AIRCRAFT WEIGHT AND BALANCE

2. The aircraft was weighed in the clean configuration, as instrumented for test, with full oil and fuel drained prior to the start of the test program. The initial weight of the aircraft was 12,074 pounds with the longitudinal center of gravity (cg) located at fuselage station (FS) 215.7. The lateral and vertical cg locations were calculated to be butt line 0.8 left and water line 150.7. The aircraft was weighed before and after each performance test flight for approximately the first half of the test program because of unreliable fuel used instrumentation. All the hover, takeoff and most of the level flight performance data were obtained during this time. The unreliable fuel used problem was resloved with the installation of newly calibrated fuel flow meters. The aircraft was also weighed towards the end of the test program and was within 6 pounds of the calculated aircraft weight. The fuel cells and external sight gauges were calibrated using a scale and 50 gallon container which was filled in 10C lb increments. The measured fuel capacity of the forward and aft cells were 144 and 208 gallons, respectively. The fuel weight for each test flight was determined before and after each flight using an external sight gauge to determine the fuel volume and a hydrometer to measure the specific gravity of the fuel. Fuel was transferred between fuel cells in flight to adjust the longitudinal cg.

PERFORMANCE

General

3. Helicopter performance was generalized through the use of nondimensional coefficients as follows:

a. Coefficient of power (C_P):

$$C_P = \frac{SHP\ (550)}{\rho A(\Omega R)^3} \qquad (1)$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{\text{GW + CABLE TENSION}}{\rho A(\Omega R)^2} \qquad (2)$$

c. Advance Ratio (μ)

$$\mu = \frac{V_T(1.6878)}{\Omega R} \qquad (3)$$

Where:

SHP = Engine output shaft horsepower (both engines)

ρ = Ambient air density (lb-sec^2/ft^4)

A = Main rotor disc area = 1809.56 ft^2

Ω = Main rotor angular velocity = 30.26 radians/sec (at 289 RPM)

R = Main rotor radius = 24.0 ft

GW = Gross weight (lb)

V_T = True airspeed (kt) = $\dfrac{V_E}{1.6878\sqrt{\rho/\rho_o}}$

1.6878 = Conversion factor (ft/sec)/kt

ρ_o = 0.0023769 (lb-sec^2/ft^4)

V_E = Equivalent airspeed (ft/sec) =

$$V_E = \left[\frac{7 \times 70.7262\, P_a}{\rho_o}\left(\left(\frac{Q_c}{P_a} + 1\right)^{2/7} - 1\right)\right]^{1/2}$$

70.7262 = Conversion factor (lb/ft^2/in. Hg)

Q_c = Dynamic pressure (in. Hg)

P_a = Ambient air pressure (in. Hg)

For a rotor speed of 289 RPM the following constants were used:

ΩR = 726.34 ft/sec

$A(\Omega R)^2 = 9.54657879 \times 10^8$ ft^4/sec^2

$A(\Omega R)^3 = 6.934025959 \times 10^{11}$ ft^5/sec^3

4. The engine output shaft torque was determined by use of the engine torque sensor. The power turbine shaft contains a torque sensor tube that mechanically displays the total twist of the shaft. A concentric reference shaft is secured by a pin at the front end of the power turbine drive shaft and is free to rotate relative to the power turbine drive shaft at the rear end. The relative rotation is due to transmitted torque, and the resulting phase angle between the reference teeth on the two shafts is picked up by the torque sensor. These torque sensors were calibrated in a test cell by the engine manufacturer and the results of this calibration are presented in figures 1 and 2. The output from the engine torque sensor was recorded on the on-board data recording system. The output SHP was determined from the engine output shaft torque and rotational speed by the following equation:

$$SHP = \frac{2\pi(N_p)Q}{33{,}000} \qquad (4)$$

Where:

Q = Engine output shaft torque (ft-lb)

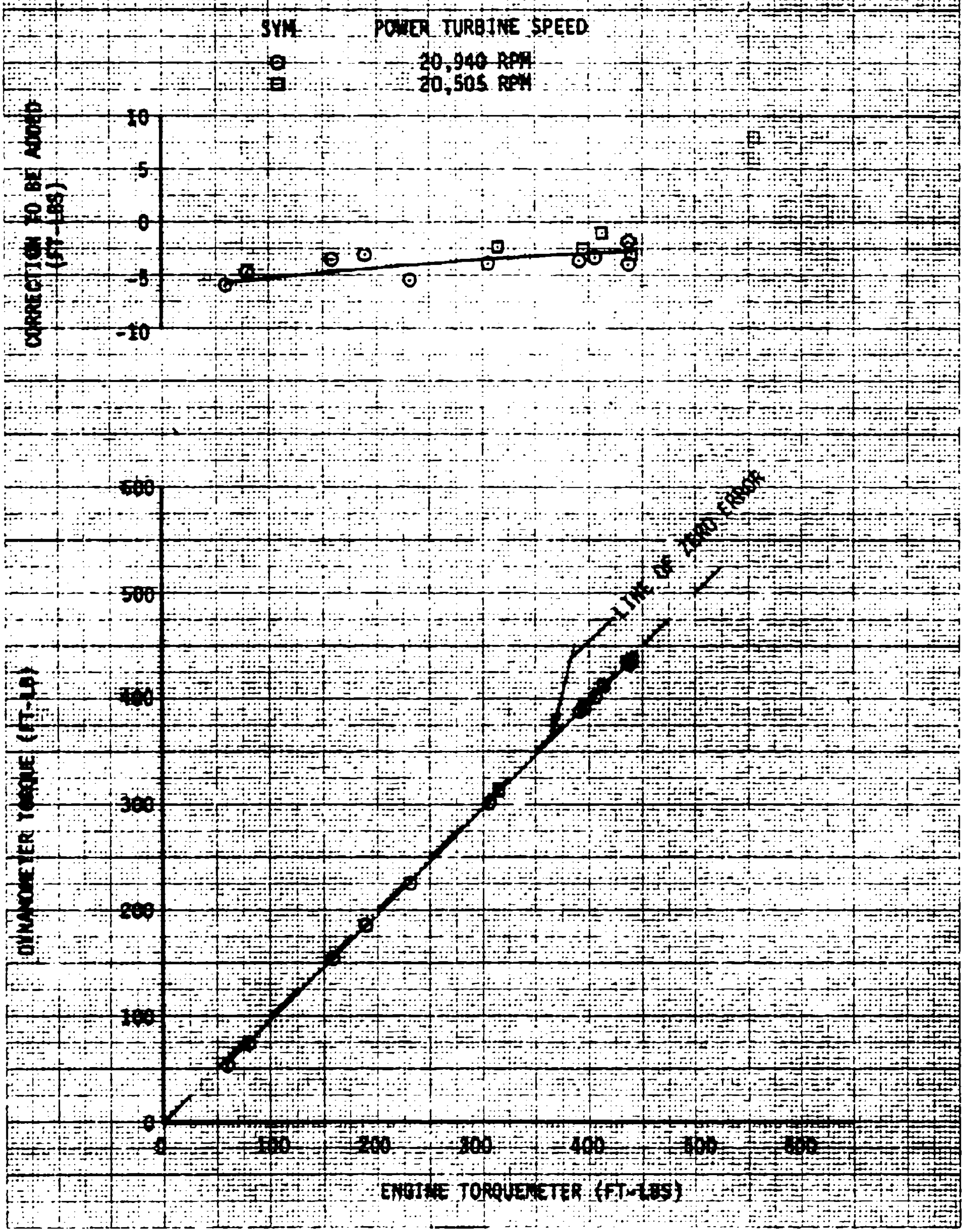
N_p = Engine output shaft rotational speed (RPM)

33,000 = Conversion factor (ft-lb/min)/SHP

SHAFT HORSEPOWER AVAILABLE

5. Shaft horsepower available for the YT700-GE-701 engine installed in the YAH-64 was obtained from data received from the United States Army Aviation Research and Development Command (AVRADCOM) (ref 15, app A). This data was calculated using the

FIGURE I
ENGINE TORQUEMETER CALIBRATION
T700-GE-701 S/N 374001



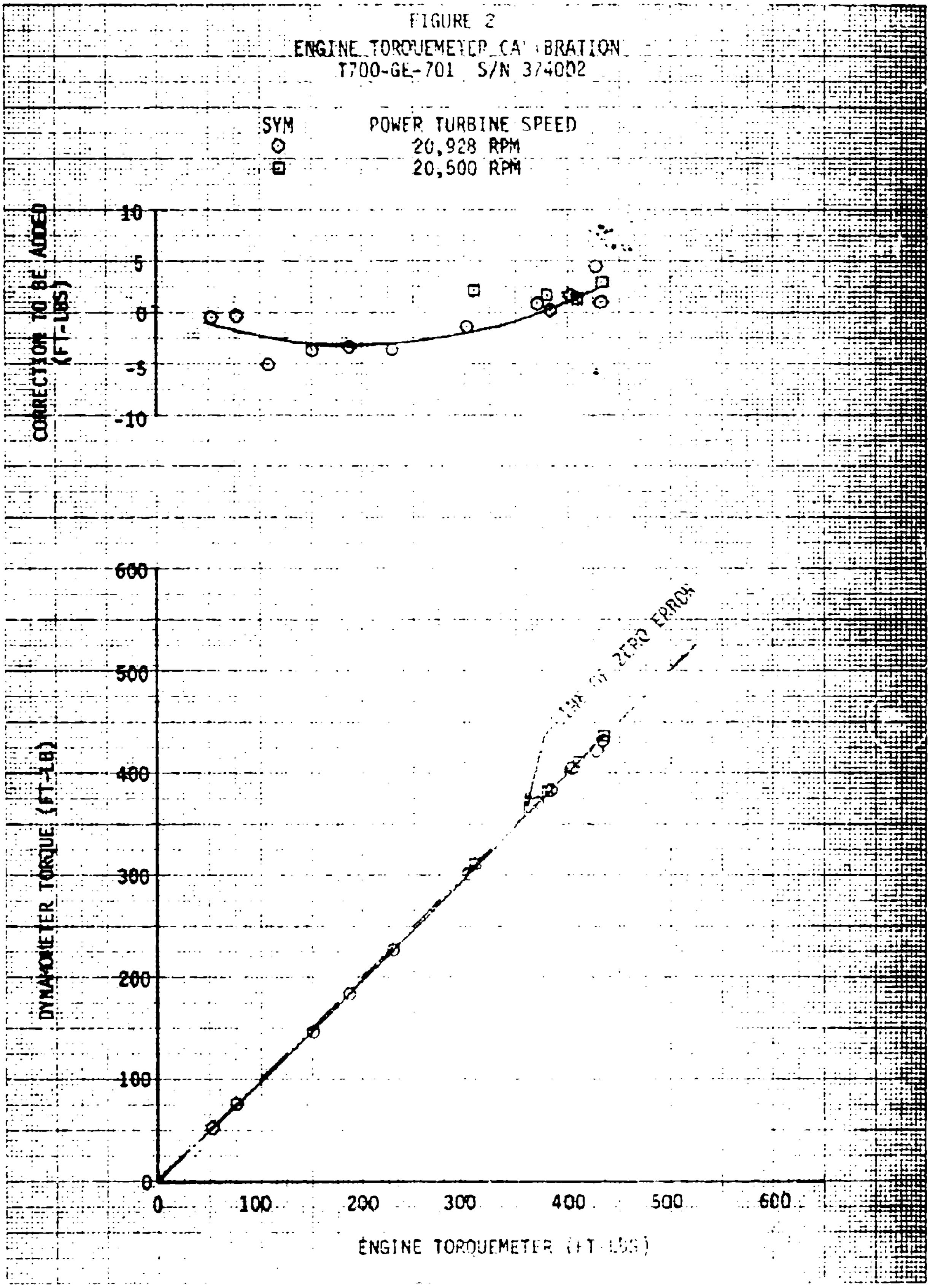


FIGURE 2
ENGINE TORQUEMETER CALIBRATION
T700-GE-701 S/N 374002

General Electric Engine Deck Number 81067 dated 4 May 1982 with a power turbine speed of 20,952 RPM. The installation losses used were based on a 1.0°C engine inlet temperature rise in a hover. In forward flight an inlet ram recovery ratio equal to 0.9957 was used and an adiabatic temperature rise at the inlet referenced to a zero degree rise in a hover was assumed. These data are presented in figures 3 through 7 and were used to determine the compliance with the system specification (ref 11, app A) for vertical climb and level flight performance. Although the power available data is the most recent, the engine deck used has not been formally approved by AVRADCOM.

Hover Performance

6. Hover performance was obtained by the tethered hover technique. Additional free flight data were obtained to add confidence to the tethered hover data. All hover tests were conducted in winds of less than 3 knots. The tethered hover technique consisted of restraining the helicopter to the ground by a cable in series with a load cell. An increase in cable tension, measured by the load cell, was equivalent to an increase in gross weight. Free-flight hover tests consisted of stabilizing the helicopter at a desired height using the radar altimeter as a height reference. Atmospheric pressure, temperature, and wind velocity were recorded from a ground weather station. All hovering data were reduced to nondimensional parameters of C_P and C_T (equations 1 and 2, respectively), and grouped according to wheel height and average density altitude. A line was faired through each set of data. Summary hovering performance was then calculated from these nondimensional plots using the power available from figure 4.

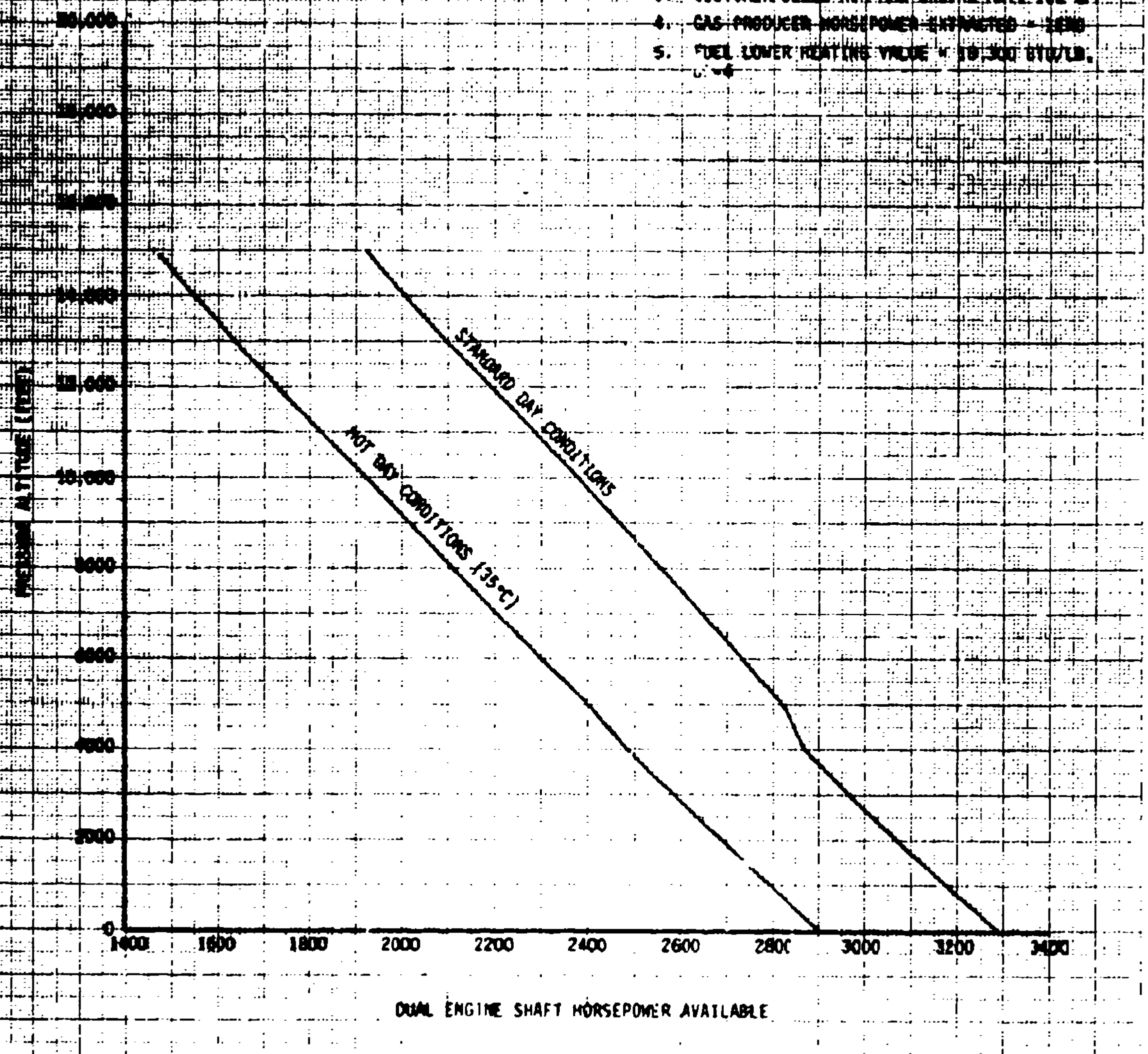
Takeoff Performance

7. Takeoff performance was determined using the level acceleration technique (para 10, Results and Discussion). Takeoff tests were conducted to obtain data of climb-out airspeed versus distance required to clear a 50-foot obstacle. These data were obtained by conducting a series of takeoffs using various climb-out airspeeds. During each series, ballast was added as necessary to maintain the desired excess power available as fuel was consumed and ambient temperature varied. A Fairchild Flight Analyzer was used to produce a photographic record of time and horizontal and vertical distances for each takeoff. The climb-out airspeed ranged from approximately 20 to 55 knots true airspeed. All takeoff tests were performed in winds of 3 knots or less.

FIGURE 3
INTERMEDIATE RATED (30 MINUTE) POWER AVAILABLE
YAH-64 USA S/N 77-23258   YT700-GE-701 ENGINE
20,952 RPM OUTPUT SHAFT SPEED (289 RPM ROTOR) ZERO KNOTS TRUE AIRSPEED

NOTE: BASED ON YT700-GE-701 ENGINE SPECIFICATION PROGRAM NUMBER 81087, CORRECTED FOR THE FOLLOWING INSTALLATION CONDITIONS:

1. INLET TEMPERATURE RISE = 1.0°C
2. EXHAUST CHARACTERISTICS OBTAINED FROM FIGURE 50, APP IV, SYSTEM SPECIFICATION FOR AAH-64.
3. CUSTOMER BLEED AIR AND ENGINE ANTI-ICE OFF
4. GAS PRODUCER HORSEPOWER EXTRACTED = ZERO
5. FUEL LOWER HEATING VALUE = 18,300 BTU/LB.



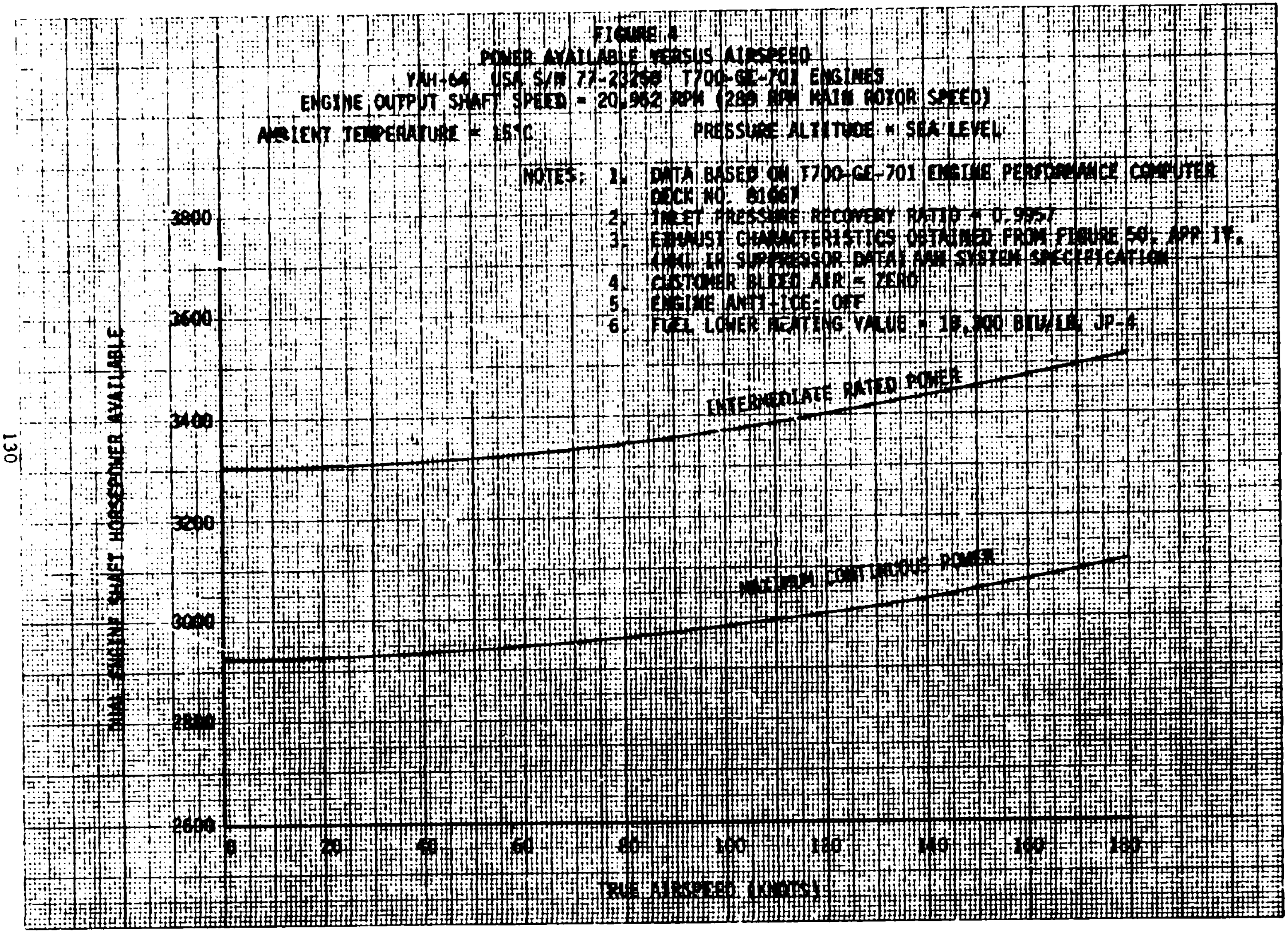


FIGURE 4
POWER AVAILABLE VERSUS AIRSPEED
YAH-64 USA S/N 77-23258 T700-GE-701 ENGINES
ENGINE OUTPUT SHAFT SPEED = 20,962 RPM (289 RPM MAIN ROTOR SPEED)
AMBIENT TEMPERATURE = 15°C
PRESSURE ALTITUDE = SEA LEVEL
NOTES: 1. DATA BASED ON T700-GE-701 ENGINE PERFORMANCE COMPUTER DECK NO. 81067
2. INLET PRESSURE RECOVERY RATIO = 0.9957
3. EXHAUST CHARACTERISTICS OBTAINED FROM FIGURE 50, APP IV, (AAH IR SUPPRESSOR DATA) AAH SYSTEM SPECIFICATION
4. CUSTOMER BLEED AIR = ZERO
5. ENGINE ANTI-ICE: OFF
6. FUEL LOWER HEATING VALUE = 18,400 BTU/LB, JP-4
INTERMEDIATE RATED POWER
MAXIMUM CONTINUOUS POWER
DUAL ENGINE SHAFT HORSEPOWER AVAILABLE
3800
3600
3400
3200
3000
2800
2600
0
20
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

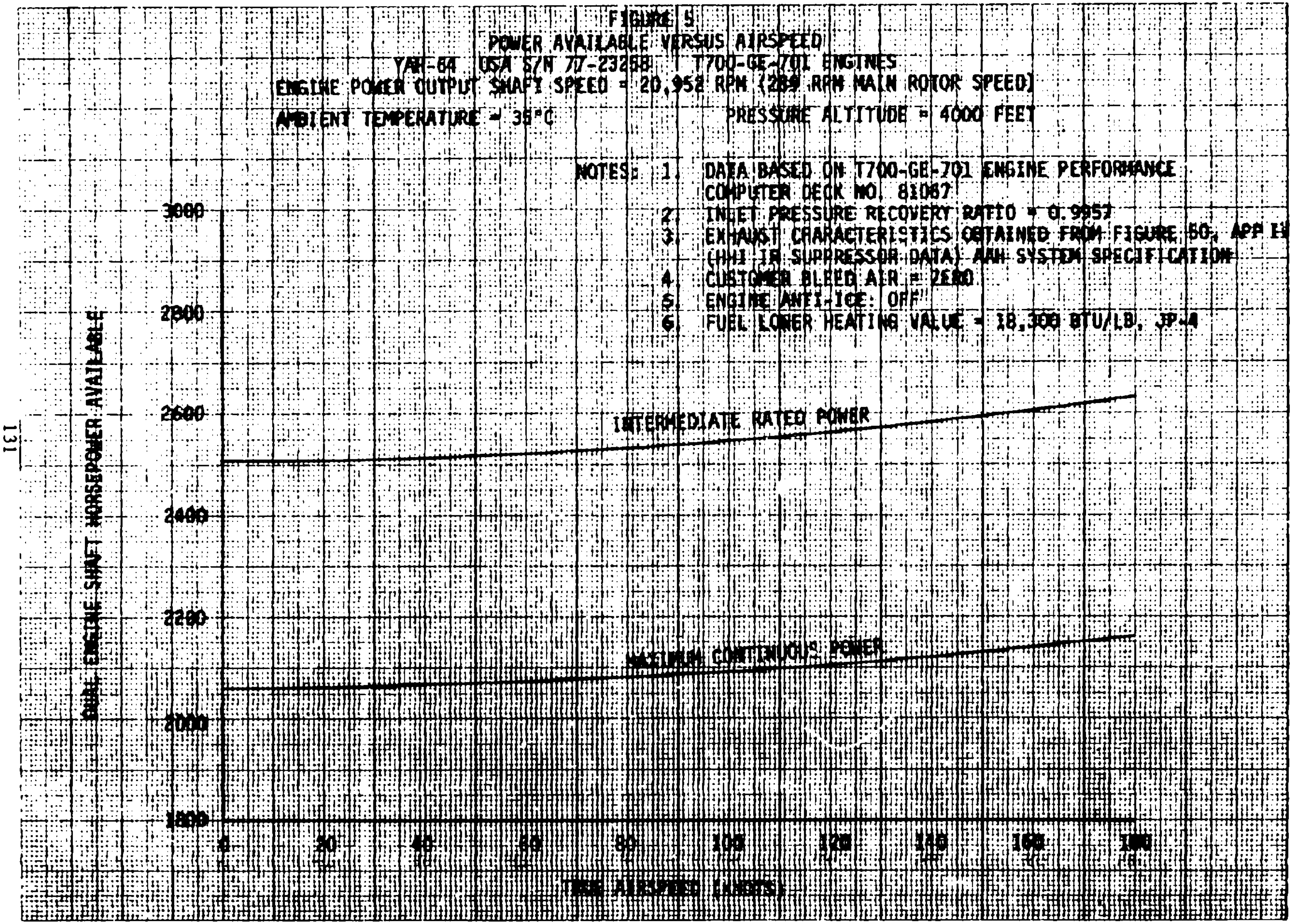
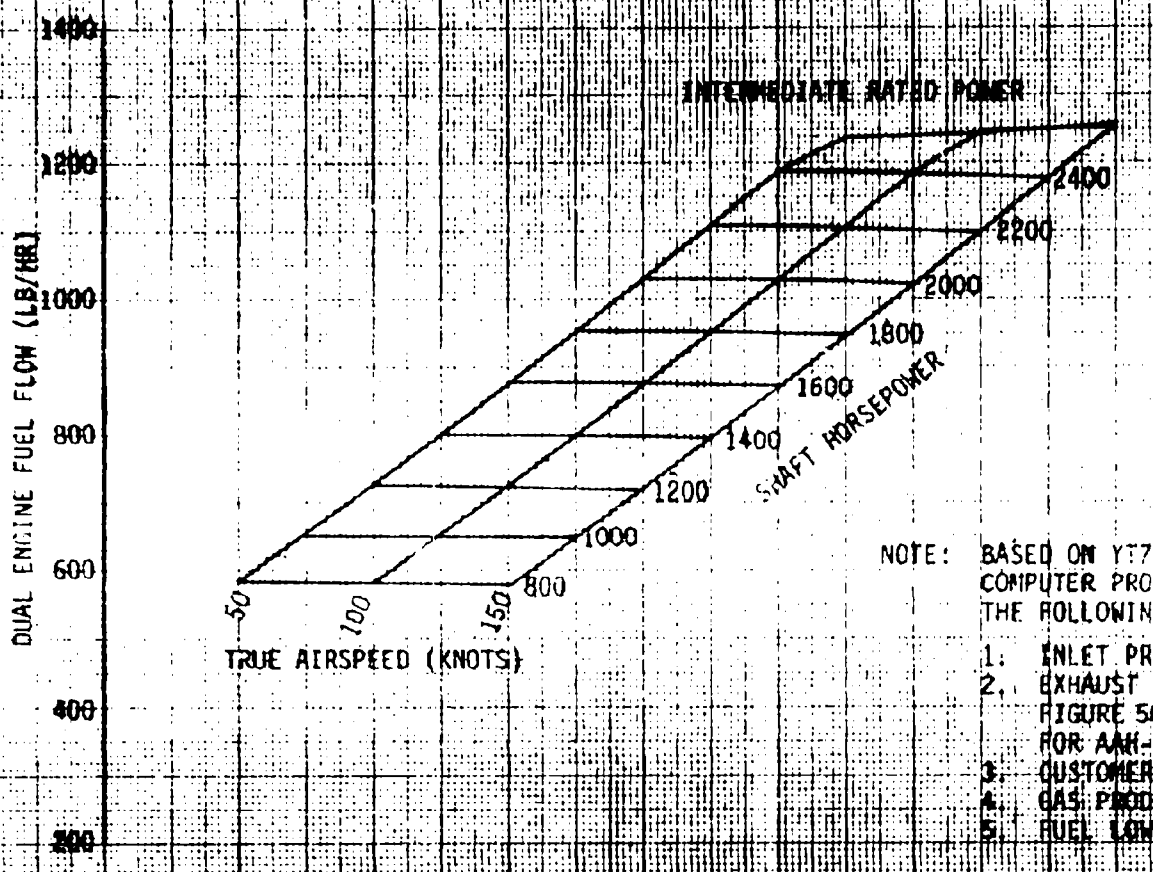


FIGURE 5
POWER AVAILABLE VERSUS AIRSPEED
YAH-64 USA S/N 77-23258 T700-GE-701 ENGINES
ENGINE POWER OUTPUT SHAFT SPEED = 20,952 RPM (289 RPM MAIN ROTOR SPEED)
AMBIENT TEMPERATURE = 35°C
PRESSURE ALTITUDE = 4000 FEET
NOTES: 1. DATA BASED ON T700-GE-701 ENGINE PERFORMANCE COMPUTER DECK NO. 81067
2. INLET PRESSURE RECOVERY RATIO = 0.9957
3. EXHAUST CHARACTERISTICS OBTAINED FROM FIGURE 50, APP II (HHI IR SUPPRESSOR DATA) AAH SYSTEM SPECIFICATION
4. CUSTOMER BLEED AIR = ZERO
5. ENGINE ANTI-ICE: OFF
6. FUEL LOWER HEATING VALUE = 18,300 BTU/LB, JP-4
INTERMEDIATE RATED POWER
MAXIMUM CONTINUOUS POWER
DUAL ENGINE SHAFT HORSEPOWER AVAILABLE
3000
2800
2600
2400
2200
2000
1800
0
20
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

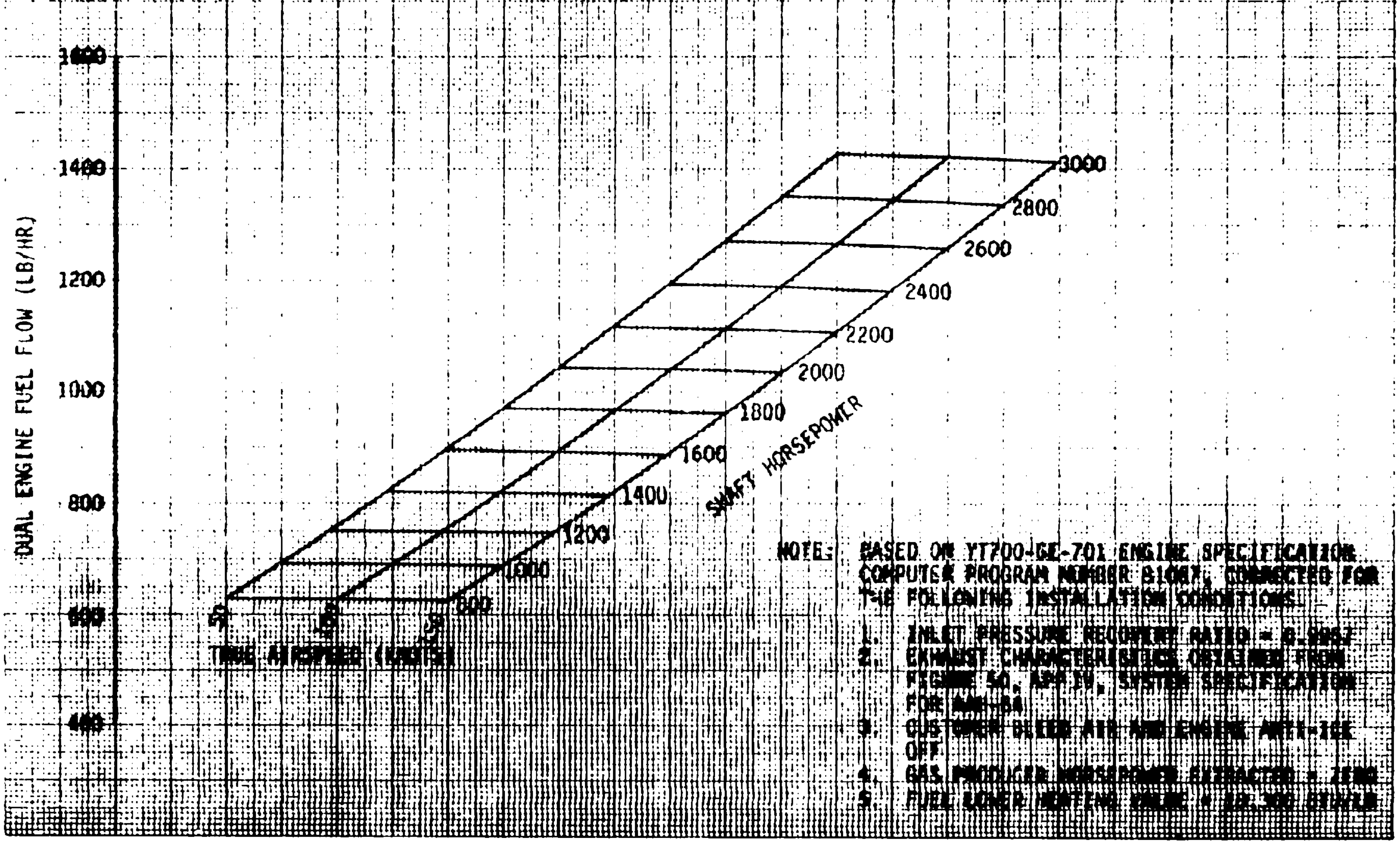
FIGURE 6
DUAL ENGINE SPECIFICATION FUEL FLOW
YAH-64 USA S/N 77-23258   YT700-GE-701 ENGINES
20,952 RPM OUTPUT SHAFT SPEED (289 RPM ROTOR) AMBIENT TEMPERATURE=35°C PRESSURE ALTITUDE=4000 FT



NOTE: BASED ON YT700-GE-701 ENGINE SPECIFICATION COMPUTER PROGRAM NUMBER 81067, CORRECTED FOR THE FOLLOWING INSTALLATION CONDITIONS.

1. INLET PRESSURE RECOVERY RATIO = 0.9967
2. EXHAUST CHARACTERISTICS OBTAINED FROM FIGURE 50, APP IV, SYSTEM SPECIFICATION FOR AAH-64
3. CUSTOMER BLEED AIR AND ENGINE ANTI ICE OFF
4. GAS PRODUCER HORSEPOWER EXTRACTED = ZERO
5. FUEL LOWER HEATING VALUE = 18,300 BTU/LB

FIGURE 7
DUAL ENGINE SPECIFICATION FUEL FLOW
YAH-64 USA S/N 77-23258   YT700-GE-701 ENGINES
20,952 RPM OUTPUT SHAFT SPEED (289 RPM ROTOR) AMBIENT TEMPERATURE=15°C PRESSURE ALTITUDE=SEA LEVEL



8. The excess power method of takeoff analysis was used. For data analysis purposes, power required was determined from the 5 foot hover curve presented in figure 2, appendix E. This hover data was obtained with no wing stores on the aircraft. The effect on power required in hover of 8 or 16 HELLFIRE missiles was considered insignificant. Power available was the average horsepower near the 50-foot wheel height point during the climb portion of the takeoff. The excess power at the 5 ft hover was then determine as follows:

$$\Delta C_P = C_{P_{\text{available}}} - C_{P_{\text{required at 5 ft hover}}}$$

The takeoff data were combined three dimensionally as distance required to clear a 50-foot obstacle versus ΔC_P and climb-out true airspeed. All dimensional takeoff performance was derived from this summary plot.

Forward Flight Climb Performance

9. Two series of climbs were conducted to determine the power and weight correction factors (K_P and K_W). A constant rotor speed and predetermined power and airspeed schedule were used. The climb airspeed schedule was determined by the airspeed corresponding to the minimum power required for level flight based on level flight performance data. To obtain K_P, a series of climbs was flown at a constant gross weight from 1000 up to 12,500 feet pressure altitude using various power settings. This series of climbs was corrected to a constant gross weight. For K_W another series of climbs was flown using dual engine maximum continuous power at various gross weights. This series of climbs was corrected for deviations from the aim power schedule. Power and weight correction factors were determined using the following equations.

a. Power correction factor (K_P):

$$K_P = \left[\frac{\Delta\ R/C}{\Delta SHP}\right] \times \left[\frac{GW_t}{33{,}000}\right] \qquad (5)$$

b. Weight correction factor (K_W):

$$K_W = \left[\frac{(R/C_2 - R/C_1)}{(SHP)\ (33{,}000)}\right]\left[\frac{(GW_1)\ (GW_2)}{GW_1 - GW_2}\right] \qquad (6)$$

Where:

$\frac{\Delta R/C}{\Delta SHP}$ = Change in rate of climb for a corresponding change in SHP, from R/C versus SHP curve, figure 12, appendix E.

GW_t = Test gross weight

R/C_1, GW_1 = Heavy gross weight and corresponding rate of climb from R/C versus GW curve, figure 13.

R/C_2, GW_2 = Light gross weight and corresponding rate of climb from R/C versus GW curve, figure 13.

33,000 = Conversion factor (ft-lb/min-SHP)

10. Power corrections were applied for variations in airspeed from the climb airspeed schedule. Any deviations from this minimum power airspeed were corrected by the following equation.

$$\Delta_{R/C} = -\frac{(K_P)(\Delta SHP)(33{,}000)}{GW_t} \qquad (7)$$

Where:

ΔSHP = Difference in level flight power required at test conditions between the test airspeed and climb schedule airspeed.

<u>Level Flight Performance</u>

11. Level flight performance was determined by using equations 1, 2, and 3. Each speed-power was flown at a constant C_T and rotor speed. To maintain gross weight ratio to air density ratio (W/σ) constant, altitude was increased as fuel was consumed. Test-day level flight power was corrected to standard-day conditions by assuming that the test-day dimensionless parameters, C_{P_t}, C_{T_t}, and μ were independent of atmospheric conditions.

Consequently, the standard-day dimensionless parameters C_{P_s}, C_{T_s},

and μ_s were identical to C_{P_t} , C_{T_t}, and μ_t, respectively. From equation 1, the following relationship can be derived.

$$SHP_s = SHP_t \frac{\rho_s}{\rho_t} \tag{8}$$

Where:

t = Test day
s = Standard day

12. Curves defined by the power required as a function of airspeed were plotted as C_P versus μ for a constant value of C_T. These curves were then joined by lines of constant μ to form a carpet plot. The reduction of this carpet plot into a family of curves C_T versus C_P, for a constant μ value allows determination of the power required as a function of airspeed for any value of C_T.

13. The specific range (NAMPP) data were derived from the test flight power required and fuel flow. The NAMPP curves were obtained from the power and airspeed from the level flight carpet plot and fuel flow from the engine model specification, corrected for installation losses, for the particular conditions. The following equation was used for determination of NAMPP.

$$NAMPP = \frac{V_T}{W_f} \tag{9}$$

Where:

V_T = True airspeed (kt)

W_f = Fuel flow (lb/hr)

The system specification endurance missions were determined based on 5 percent conservatism applied to the fuel flow.

14. Changes in the equivalent flat plate area (Δf_e) for various aircraft configurations were calculated by the following equation:

$$\Delta f_e = \frac{2(\Delta C_P)\ A}{\mu^3} \tag{10}$$

Where:

Δf_e = Change in equivalent flat plate area (ft^2)

ΔC_P = Change in power coefficient at constant C_T and μ

15. The total electrical power consumed by the test aircraft was measured with the aircraft on the ground with all test systems operational and instrumentation system ON. The instrumentation included many transducers, signal conditioning equipment, a magnetic tape recorder and telemetry equipment. The total electrical power required by the test aircraft was 1.2 kilowatts which equates to 1.6 horsepower. The environmental control unit (ENCU) was operated for all flights at a level for pilot and copilot comfort. This unit consumes a variable amount of power through the shaft driven compressor. The drag due to the aircraft configuration differences presented in table 1, appendix B were not considered in any of the data in this report. Also no corrections for the power consumed by the ENCU or electrical equipment installed in the test aircraft were made.

Autorotational Descent Performance

16. Autorotational descent performance data were acquired at various stabilized airspeeds with rotor speed held constant and at various rotor speeds at constant airspeed. The tapeline rates of descent were calculated by the following equation.

$$R/D \text{ tapeline} = \left[\frac{dH_p}{dt}\right] \times \left[\frac{T_t}{T_s}\right] \tag{11}$$

Where:

$\frac{dH_p}{dt}$ = Change in pressure altitude per given time (ft/sec)

T_t = Test ambient air temperature (°K)

T_s = Standard ambient air temperature (°K)

HANDLING QUALITIES

17. Stability and control data were collected and evaluated using standard test methods as described in reference 14, appendix A. The Handling Qualities Rating Scale presented in figure 8 was used to augment pilot comments relative to handling qualities. Turbulence reporting criteria (ref 19, app A) used during this evaluation are presented in table 1.

VIBRATION

18. The vibration data were reduced by means of a fast Fourier transform from the analog flight tape. Vibration levels, representing peak amplitudes, were extracted from this analysis at selected harmonics of the main rotor frequency. The Vibration Rating Scale, presented in figure 9, was used to augment crew comments on aircraft vibration levels.

AIRSPEED SYSTEM CALIBRATION

19. The boom airspeed system was calibrated by using a pace aircraft and by the trailing bomb method to determine the airspeed and altimeter position error. Calibrated airspeed and pressure altitude were obtained by correcting indicated values for instrument error and position error (fig. 1, app C). Altitude position error was calculated using the airspeed position error and assuming all errors were introduced at the static port using the following equation.

$$\Delta P_p = 1.4\, P_{a_o}\left(\frac{V_{ic}}{a_o}\right)\left[1 + 0.2\left(\frac{V_{ic}}{a_o}\right)^2\right]^{2.5}\left(\frac{\Delta V_{pc}}{a_o}\right) + 0.7\, P_{a_o}\left[1 + 0.2\left(\frac{V_{ic}}{a_o}\right)^2\right]^{1.5}\left[1 + 1.2\left(\frac{V_{ic}}{a_o}\right)^2\right]\left(\frac{\Delta V_{pc}}{a_o}\right)^2 \quad (3)$$

Where:

ΔP_p = Static position error

P_{a_o} = Atmospheric pressure at standard-day sea level

V_{ic} = Instrument corrected indicated airspeed

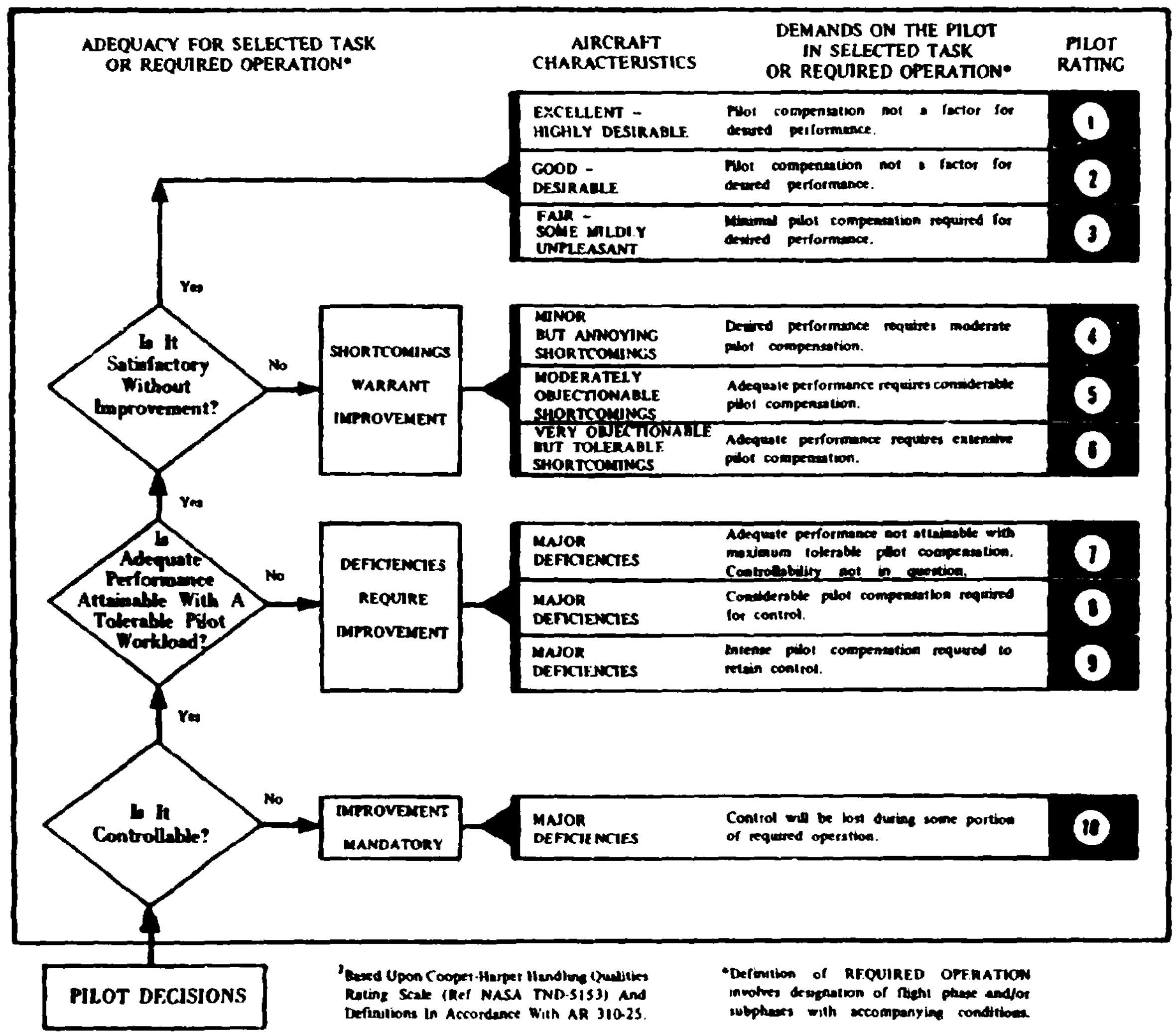
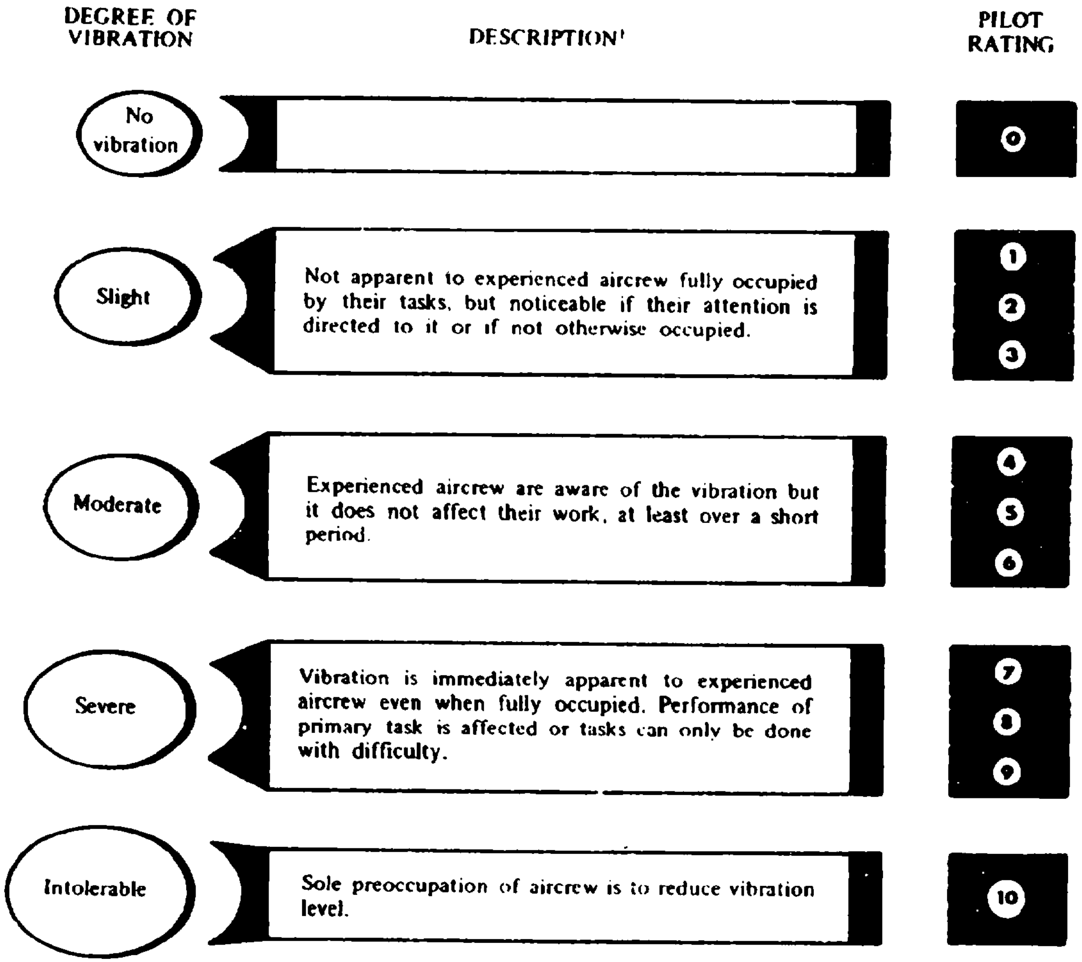


Figure 8. Handling Qualities Rating Scale

Table 1. Turbulence Reporting Criteria

Intensity	Aircraft Reaction	Reaction Inside Aircraft	Reporting Term-Definition
Light	Turbulence that momentarily causes slight erratic changes in altitude and/or attitude (pitch, roll, yaw). Report as Light Turbulence;* or Turbulence that causes slight, rapid and somewhat rhythmic bumpiness without appreciable changes in altitude or attitude. Report as Light Chop	Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects may be displaced slightly. Food service may be conducted and little or no difficulty is encountered in walking	
Moderate	Turbulence that is similar to Light Turbulene but of greater intensity. Changes in altitude and/or attitude occur but the aircraft remains in positive control at all times. It usually causes variations in indicated airspeed. Report as Moderate Turbulence.* or Turbulence that is similar to Light Chop but of greater intensity. It causes rapid bumps or jolts without appreciable changes in aircraft altitude or attitude. Report as Moderate Chop.	Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged. Food service and walking are difficult.	Occasional - Less than 1/3 of the tim Intermittent - 1/3 to 2/3 Continuous - More than 2[illegible]
Severe	Turbulence that causes large, abrupt changes in altitude and/or attitude. It usually causes large variations in indicated airspeed. Aircraft may be momentarily out of control. Report as Severe Turbulence.*	Occupants are forced violently against seat belts or shoulder straps. Unsecured objects are tossed about. Food service and walking are impossible.	
Extreme	Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage. Report as Extreme Turbulence.*		



[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 9. Vibration Rating Scale

a_o = Speed of sound at standard-day sea level

ΔV_{pc} = Measured airspeed position error.

DEFINITIONS

20. The following definitions of deficiencies and shortcomings were used during the evaluation.

a. Deficiency - A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipment's operational capability.

b. Shortcoming - An imperfection or malfunction occurring during the life cycle of equipment which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the useability of the material or end product.

APPENDIX E. TEST DATA

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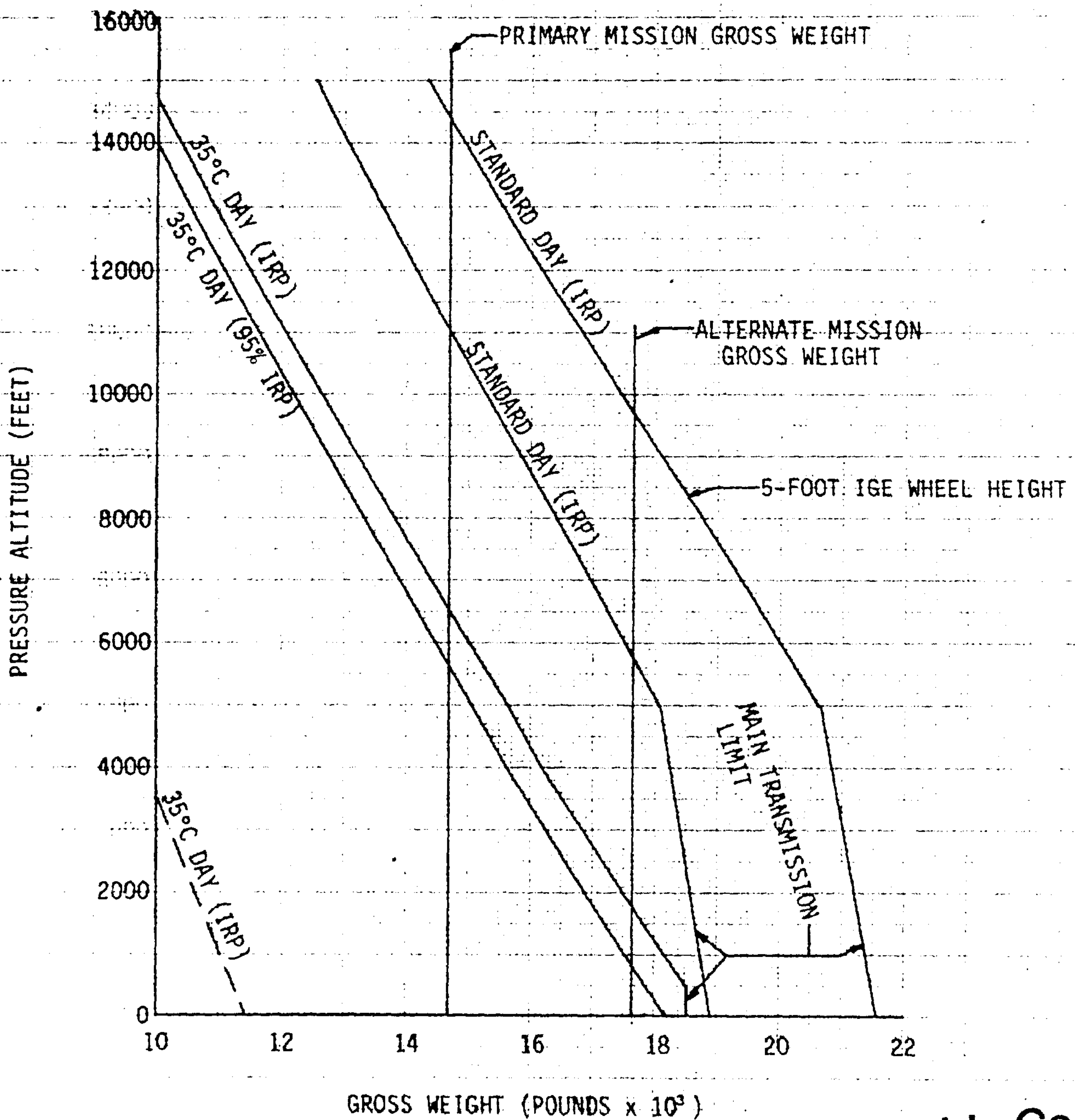
FIGURE 1

SUMMARY HOVER PERFORMANCE

YAH-64 USA S/N 77-23258

NOTES:
1. HOVER DATA DERIVED FROM FIGURES 2 AND 5
2. MAIN ROTOR SPEED 289 RPM, CLEAN CONFIGURATION
3. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
4. POWER AVAILABLE DERIVED FROM FIGURE 3, APP D.
5. WINDS 3 KNOTS OR LESS
6. DASHED LINE DENOTES SINGLE ENGINE OPERATION
7. 100-FOOT WHEEL HEIGHT EXCEPT WHERE NOTED



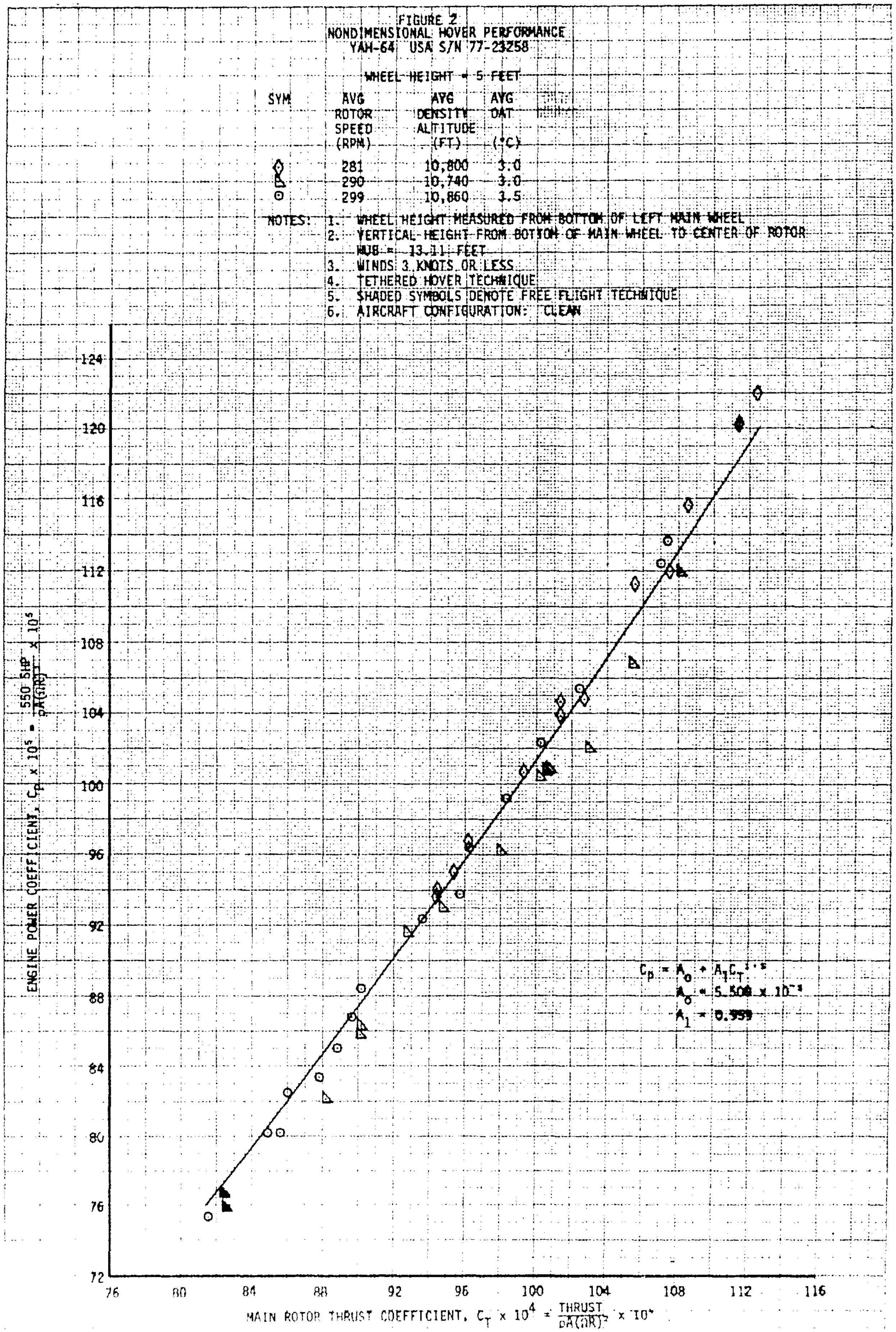


FIGURE 2
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258
WHEEL HEIGHT = 5 FEET
SYM
AVG ROTOR SPEED (RPM)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
281 10,800 3.0
290 10,740 3.0
299 10,860 3.5
NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. VERTICAL HEIGHT FROM BOTTOM OF MAIN WHEEL TO CENTER OF ROTOR HUB = 13.11 FEET
3. WINDS 3 KNOTS OR LESS
4. TETHERED HOVER TECHNIQUE
5. SHADED SYMBOLS DENOTE FREE FLIGHT TECHNIQUE
6. AIRCRAFT CONFIGURATION: CLEAN
ENGINE POWER COEFFICIENT, $C_P \times 10^5 = \frac{550\ SHP}{\rho A(\Omega R)^3} \times 10^5$
MAIN ROTOR THRUST COEFFICIENT, $C_T \times 10^4 = \frac{THRUST}{\rho A(\Omega R)^2} \times 10^4$
$C_P = A_0 + A_1 C_T^{1.5}$
$A_0 = 5.500 \times 10^{-5}$
$A_1 = 0.959$
124
120
116
112
108
104
100
96
92
88
84
80
76
72
76
80
84
88
92
96
100
104
108
112
116

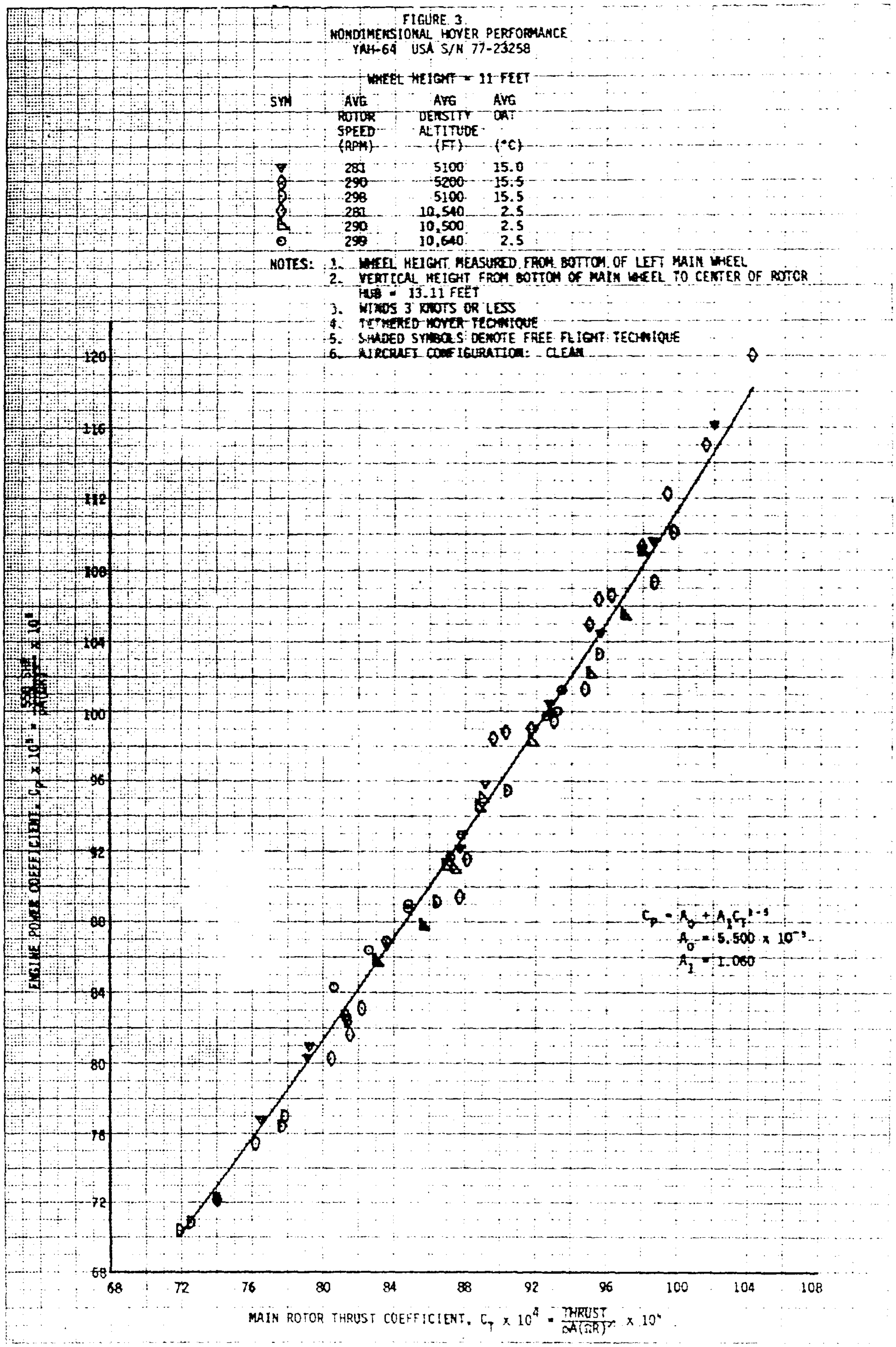
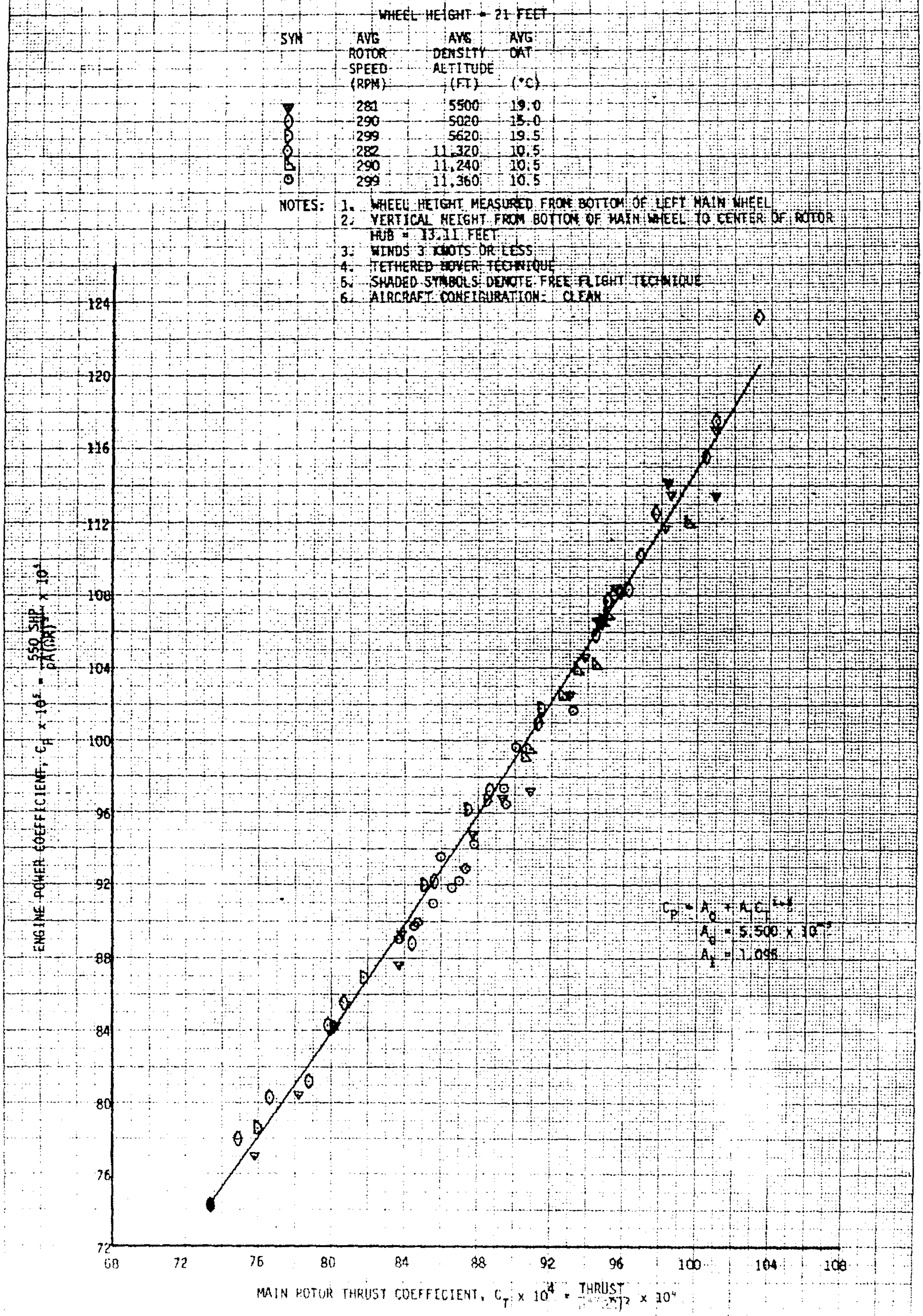


FIGURE 3
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258

FIGURE 4
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258



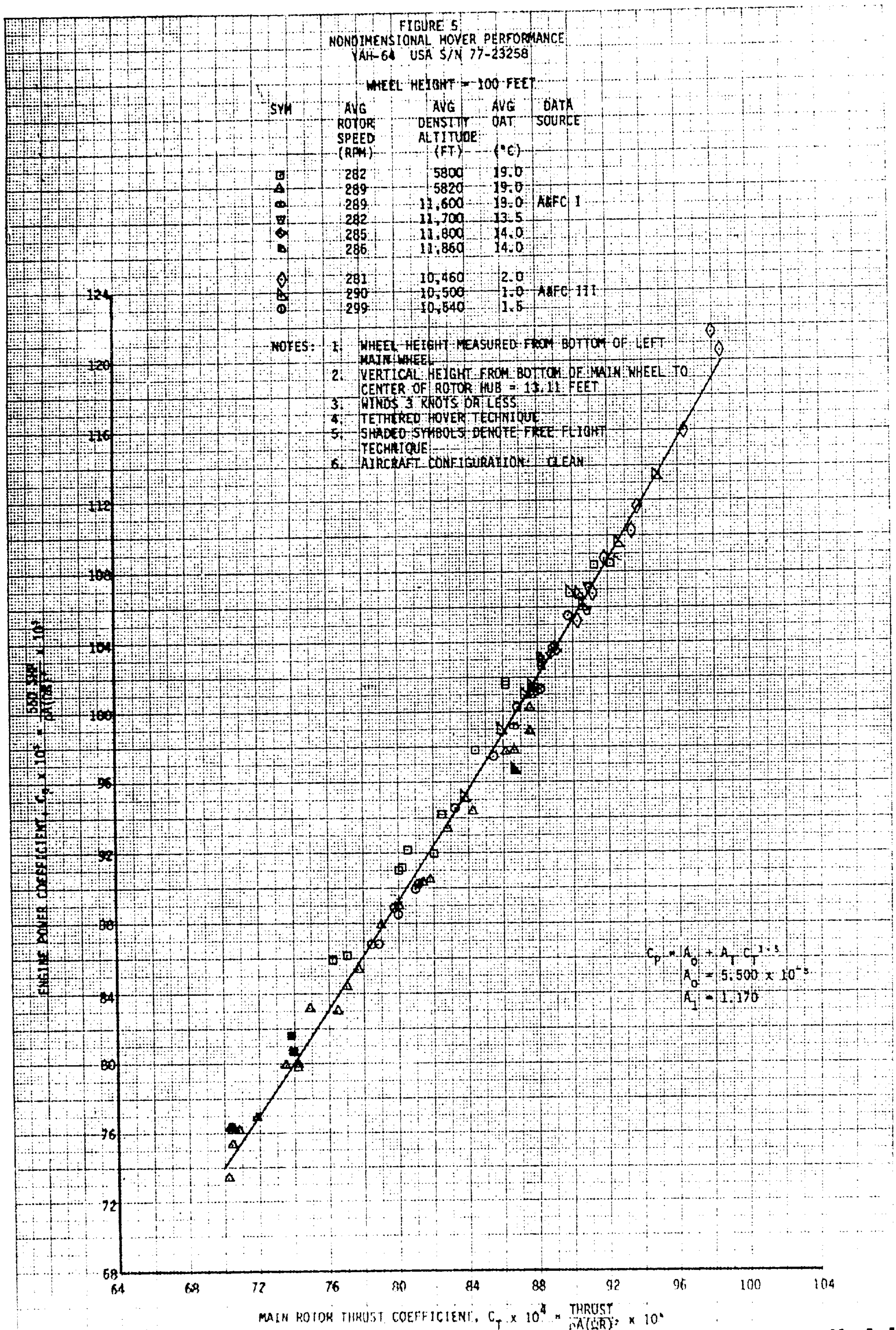


FIGURE 5
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258
WHEEL HEIGHT = 100 FEET
SYM
AVG ROTOR SPEED (RPM)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
DATA SOURCE
282 5800 19.0
289 5820 19.0
289 11,600 19.0 A&FC I
282 11,700 13.5
285 11,800 14.0
286 11,860 14.0
281 10,460 2.0
290 10,500 1.0 A&FC III
299 10,540 1.5
NOTES:
1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. VERTICAL HEIGHT FROM BOTTOM OF MAIN WHEEL TO CENTER OF ROTOR HUB = 13.11 FEET
3. WINDS 3 KNOTS OR LESS
4. TETHERED HOVER TECHNIQUE
5. SHADED SYMBOLS DENOTE FREE FLIGHT TECHNIQUE
6. AIRCRAFT CONFIGURATION: CLEAN
$C_P = A_0 + A_1 C_T^{1.5}$
$A_0 = 5.500 \times 10^{-5}$
$A_1 = 1.170$
ENGINE POWER COEFFICIENT, $C_P \times 10^5 = \frac{550\ SHP}{\rho A (\Omega R)^3} \times 10^5$
124
120
116
112
108
104
100
96
92
88
84
80
76
72
68
64
68
72
76
80
84
88
92
96
100
104
MAIN ROTOR THRUST COEFFICIENT, $C_T \times 10^4 = \frac{THRUST}{\rho A (\Omega R)^2} \times 10^4$

FIGURE 6
TAKEOFF PERFORMANCE SUMMARY
YAH-64 USA S/N 77-23258

NOTES:
1. LEVEL ACCELERATION FROM A 5-FOOT HOVER TECHNIQUE USED
2. CURVES DERIVED FROM FIGURES 7 THROUGH 10
3. ΔC_P = MAXIMUM AVAILABLE C_P AT 5-FOOT WHEEL HEIGHT HOVER MINUS THE C_P REQUIRED TO HOVER AT 5 FEET, TEST DAY CONDITIONS
4. MAIN ROTOR SPEED MAINTAINED AT 289 RPM
5. WINDS 3 KNOTS OR LESS
6. AIRCRAFT CONFIGURATION: 8-HELLFIRE AND TWO 19-ROUND ROCKET PODS
7. DASHED LINES DENOTE INTERPOLATED DATA

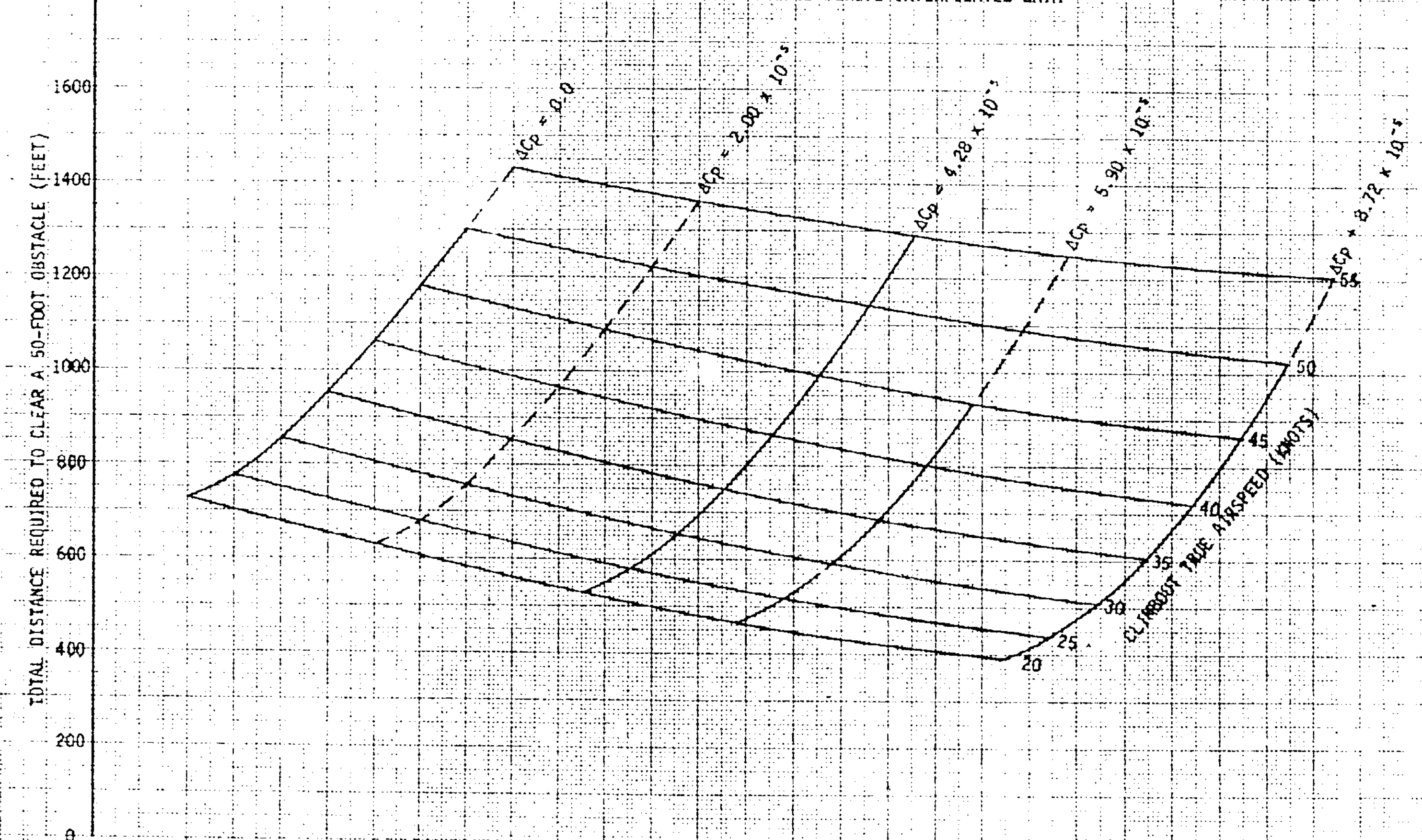


FIGURE 7
TAKEOFF PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_P
16,640	202.5(FWD)	0.8(LT)	10,960	7.5	290	0.0

NOTES: 1. 8-HELLFIRE, TWO 19-ROUND ROCKET POD CONFIGURATION
2. DATA OBTAINED USING A LEVEL ACCELERATION FROM A 5-FOOT HOVER TECHNIQUE
3. WINDS 3 KNOTS OR LESS

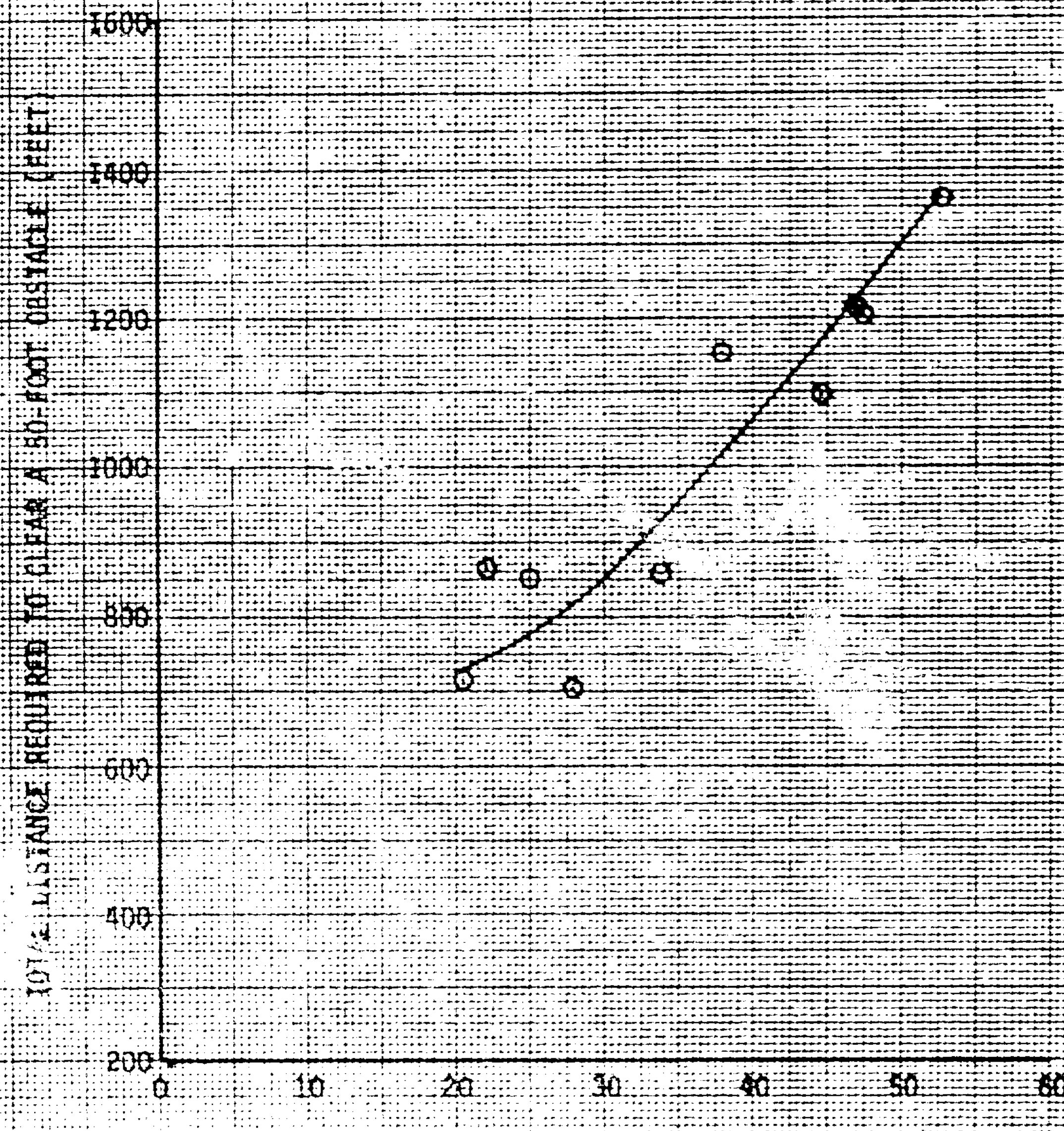


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FIGURE 8
TAKEOFF PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG ΔC_P
16,440	202.5(FWD)	0.8(LT)	10,720	7.5	290	4.28×10^{-5}

NOTE: 1. 8-HELLFIRE, TWO 19-ROUND ROCKET POD CONFIGURATION
2. DATA OBTAINED USING A LEVEL ACCELERATION FROM A 5-FOOT HOVER TECHNIQUE
3. WINDS 3 KNOTS OR LESS

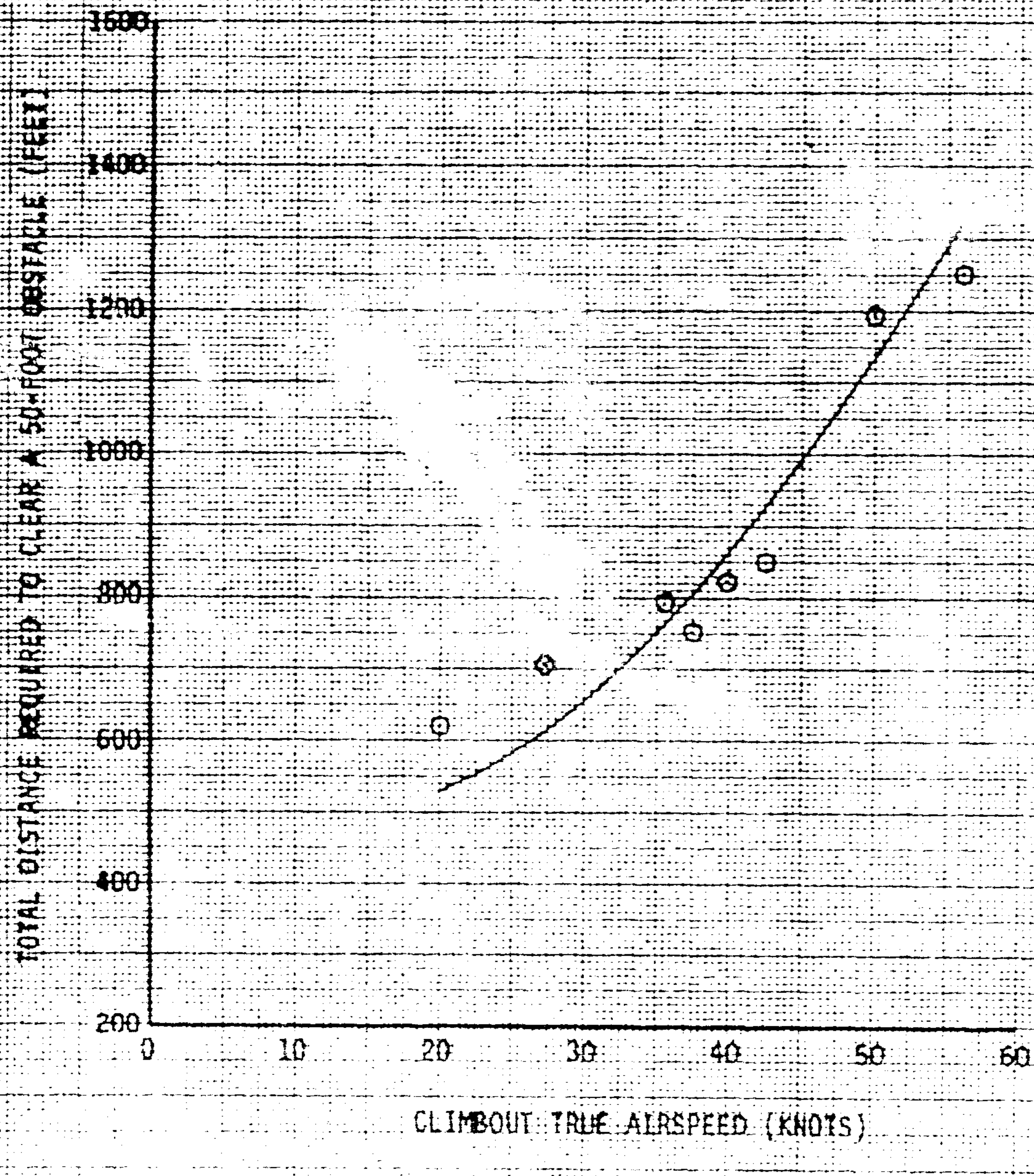


FIGURE 9
TAKEOFF PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_P
15,900	202.7(FWD)	0.8(LT)	11,240	8.5	290	5.90×10^{-5}

NOTES: 1. 8-HELLFIRE, TWO 19-ROUND ROCKET POD CONFIGURATION
2. DATA OBTAINED USING A LEVEL ACCELERATION FROM A 5-FOOT HOVER TECHNIQUE
3. WINDS 3 KNOTS OR LESS

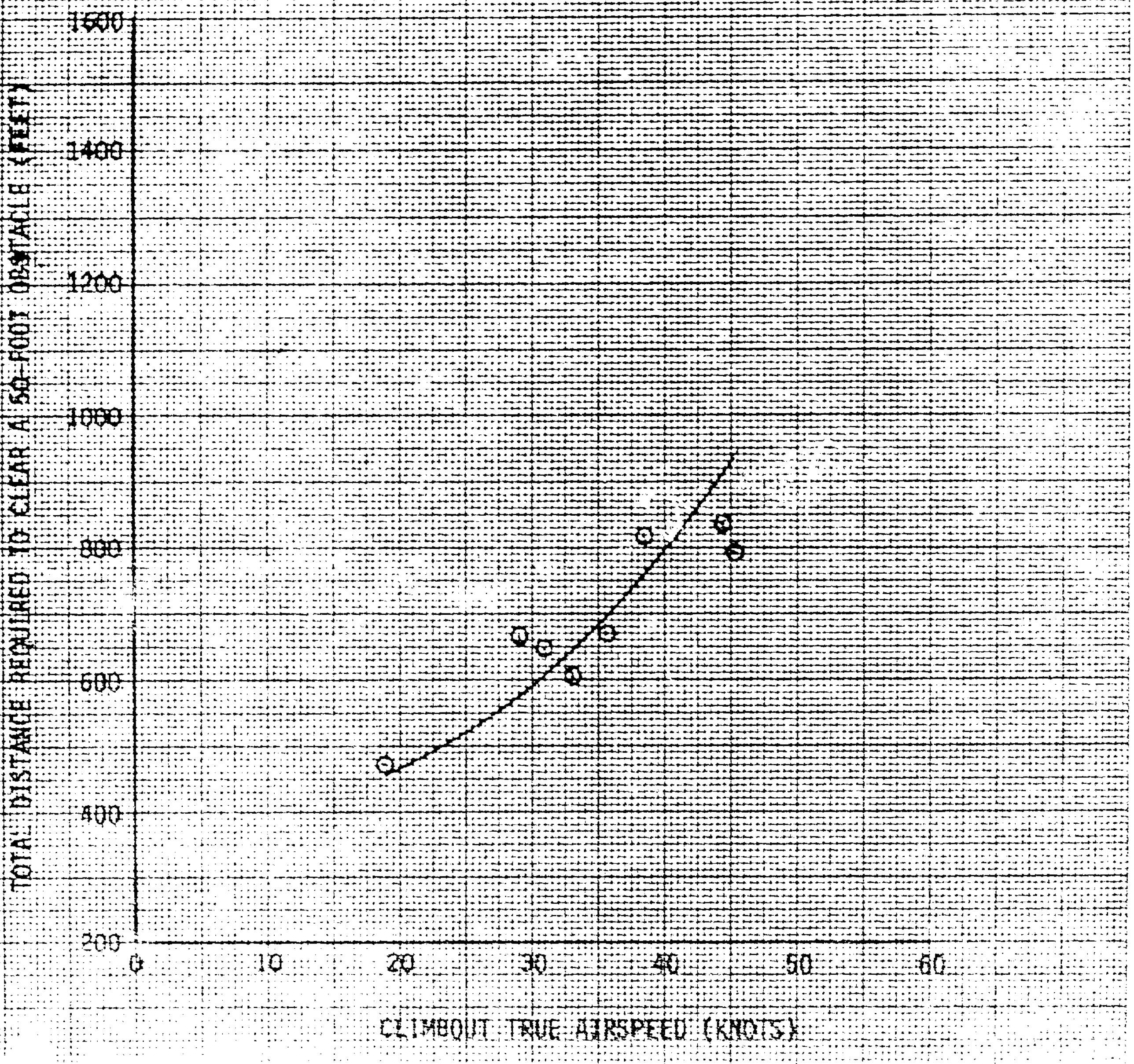
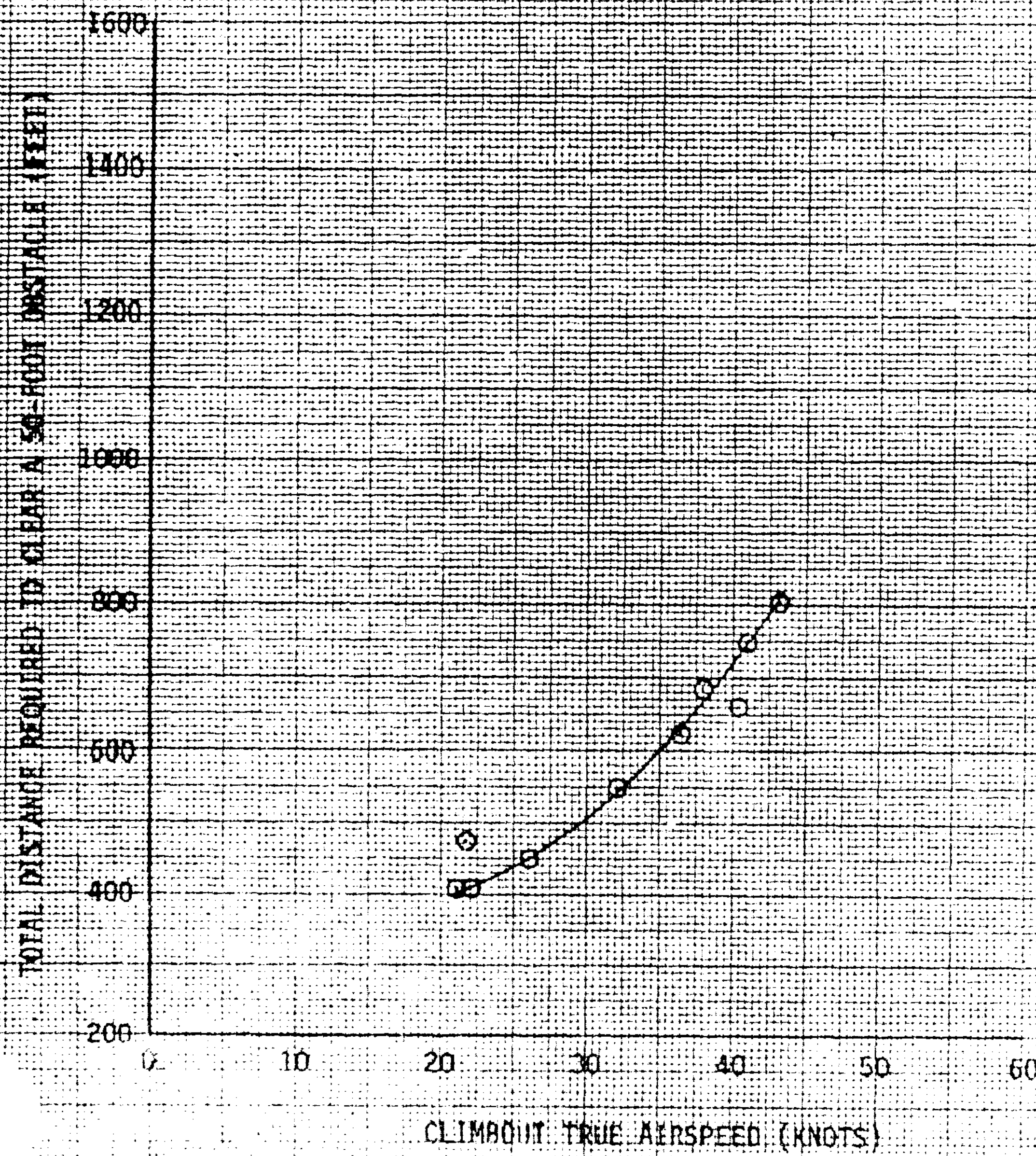


FIGURE 10
TAKEOFF PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG ΔC_P
15,800	202.5(FWD)	0.8(LT)	10,940	10.0	289	8.72×10^{-5}

NOTES:
1. 8-HELLFIRE, TWO 19-ROUND ROCKET POD CONFIGURATION
2. DATA OBTAINED USING A LEVEL ACCELERATION FROM A 5-FOOT HOVER TECHNIQUE
3. WINDS 3 KNOTS OR LESS



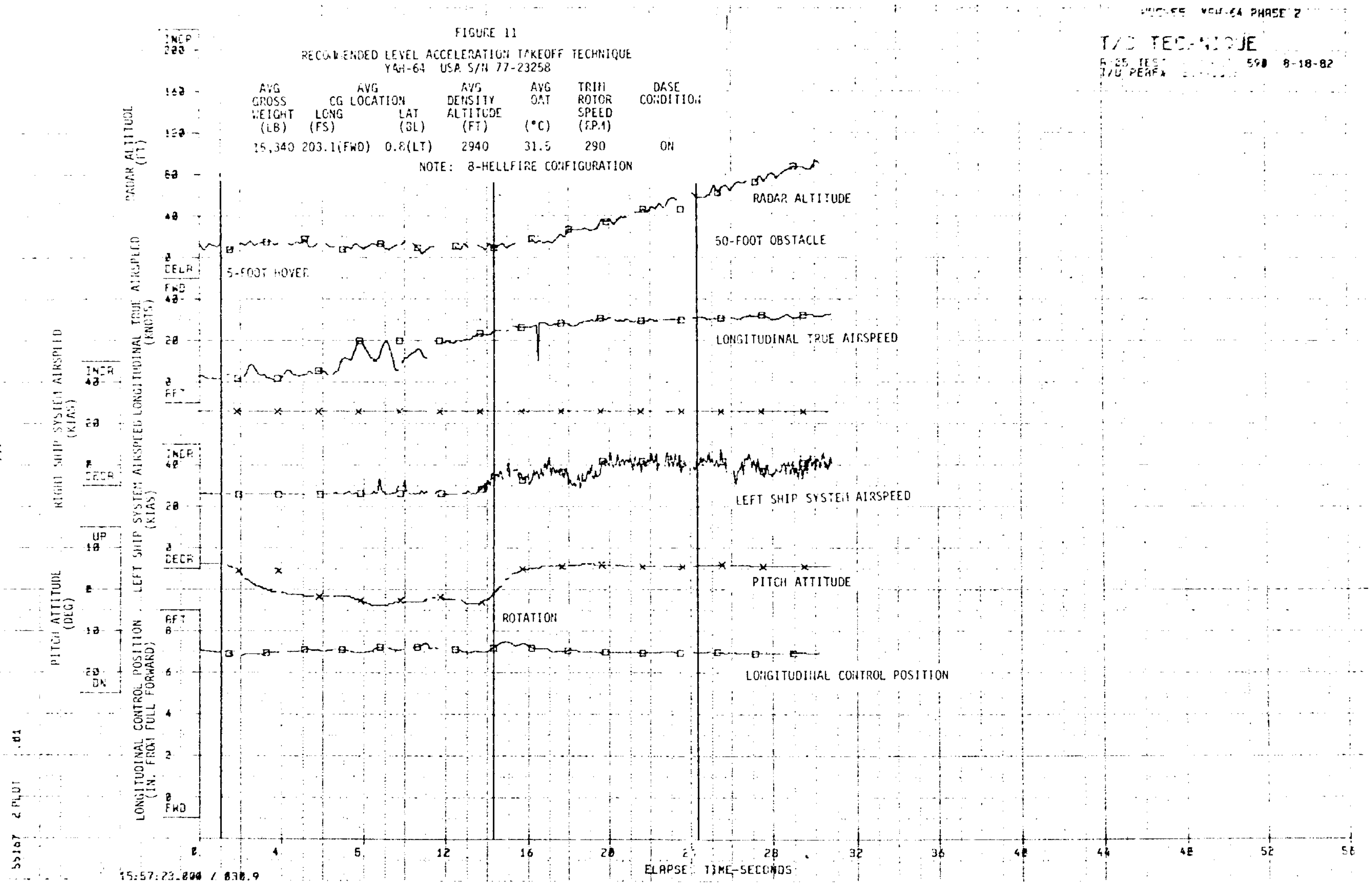


FIGURE 11
RECOMMENDED LEVEL ACCELERATION TAKEOFF TECHNIQUE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	DASE CONDITION
15,340	203.1(FWD)	0.8(LT)	2940	31.5	290	ON

NOTE: 8-HELLFIRE CONFIGURATION

FIGURE 12
VERTICAL CLIMB PERFORMANCE SUMMARY
YAH-64 USA S/N 77-23258

GROSS WEIGHT = 14,694 LB
AMBIENT TEMPERATURE = 35°C
ROTOR SPEED = 289 RPM

8-HELLFIRE CONFIGURATION

NOTES: 1. POWER AVAILABLE OBTAINED FROM FIGURE 3, APPENDIX D.
2. DATA DERIVED FROM FIGURE 1 APPENDIX E AND FIGURE 2, APPENDIX E, FINAL REPORT, A&FC, PART 2.

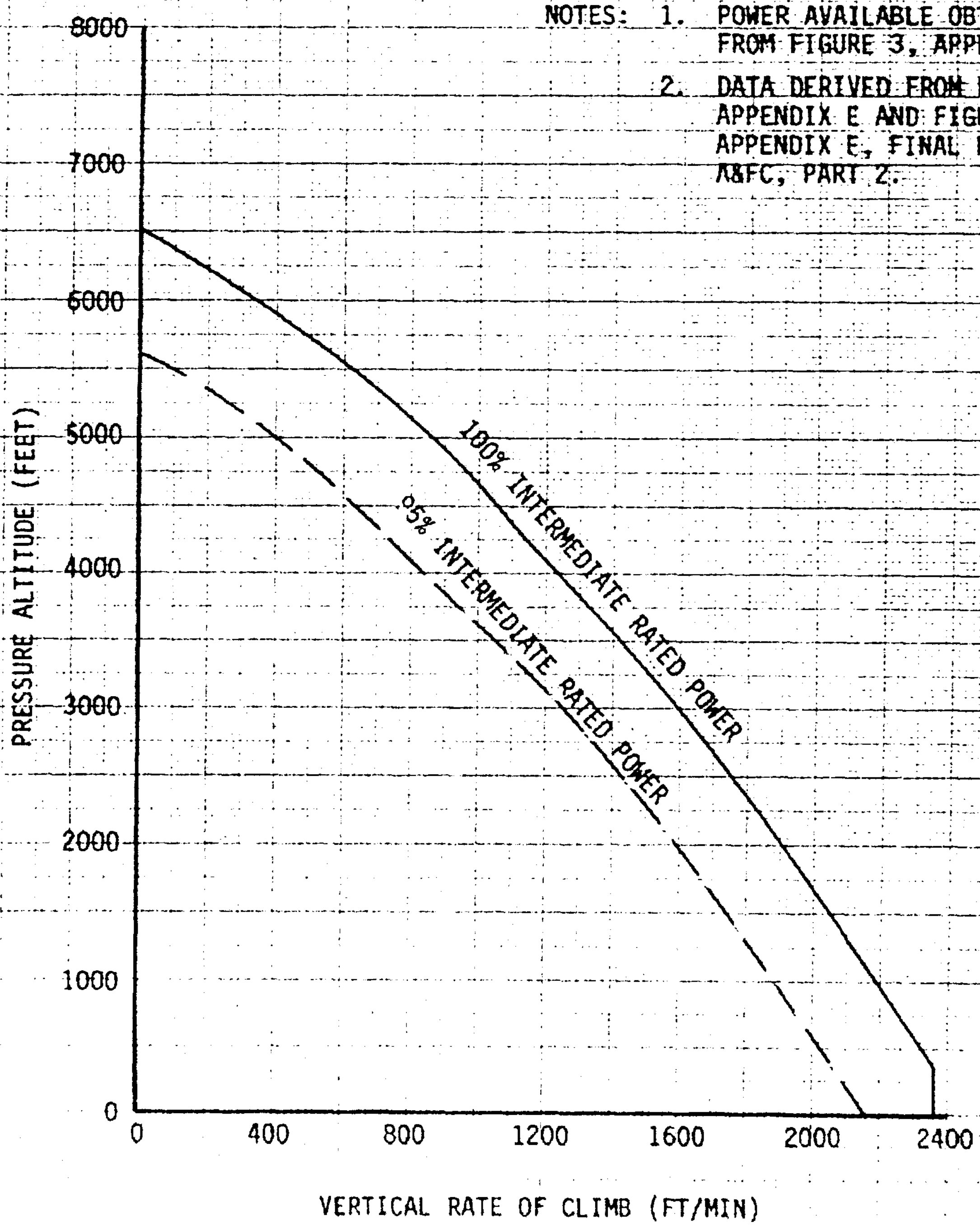


FIGURE 13
VARIATION IN RATE OF CLIMB AS A FUNCTION OF SHAFT HORSEPOWER
YAH-64 USA S/N 77-23258

SYMBOL	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)
⊙	2500	15,550	290
⊡	5000	15,550	290
◊	7500	15,550	290
◺	10,000	15,550	290
Δ	12,500	15,550	290

NOTES:
1. 8-HELLFIRE CONFIGURATION.
2. TESTS CONDUCTED AT BEST RATE OF CLIMB SPEED AS DETERMINED FROM FIGURES 15 THROUGH 17.
3. $K_p = \frac{\Delta R/C}{\Delta SHP} X \frac{GW}{33,000} = 0.77$

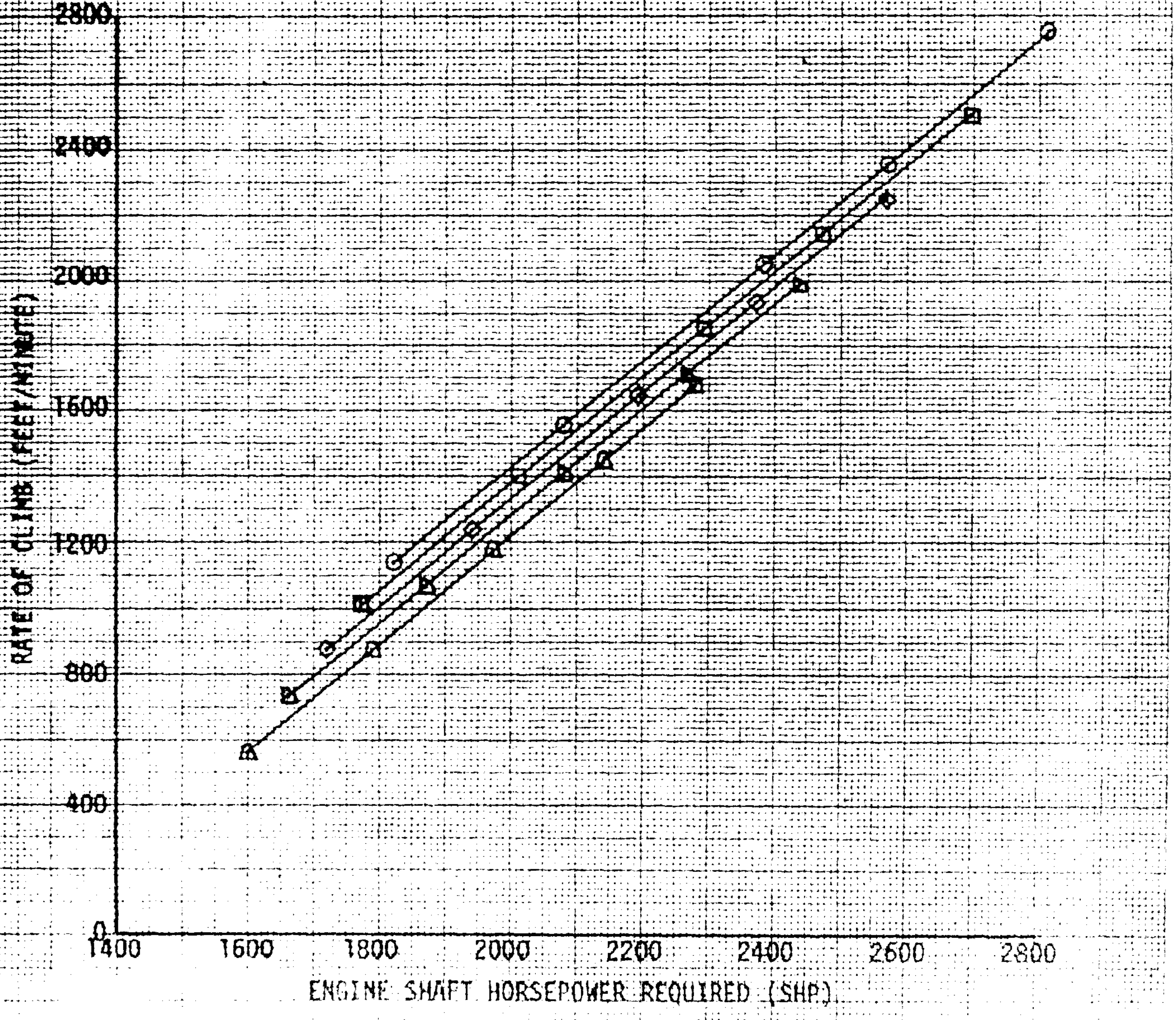


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FIGURE 14
VARIATION IN RATE OF CLIMB AS A FUNCTION OF THRUST COEFFICIENT
YAH-64 USA S/N 77-23258

SYMBOL	DENSITY ALTITUDE (FT)	ROTOR SPEED (RPM)	SHAFT HORSEPOWER (SHP)
○	2500	290	2490
□	5000	290	2307
◇	7500	290	2181
◺	10,000	290	2078
△	12,500	290	1960

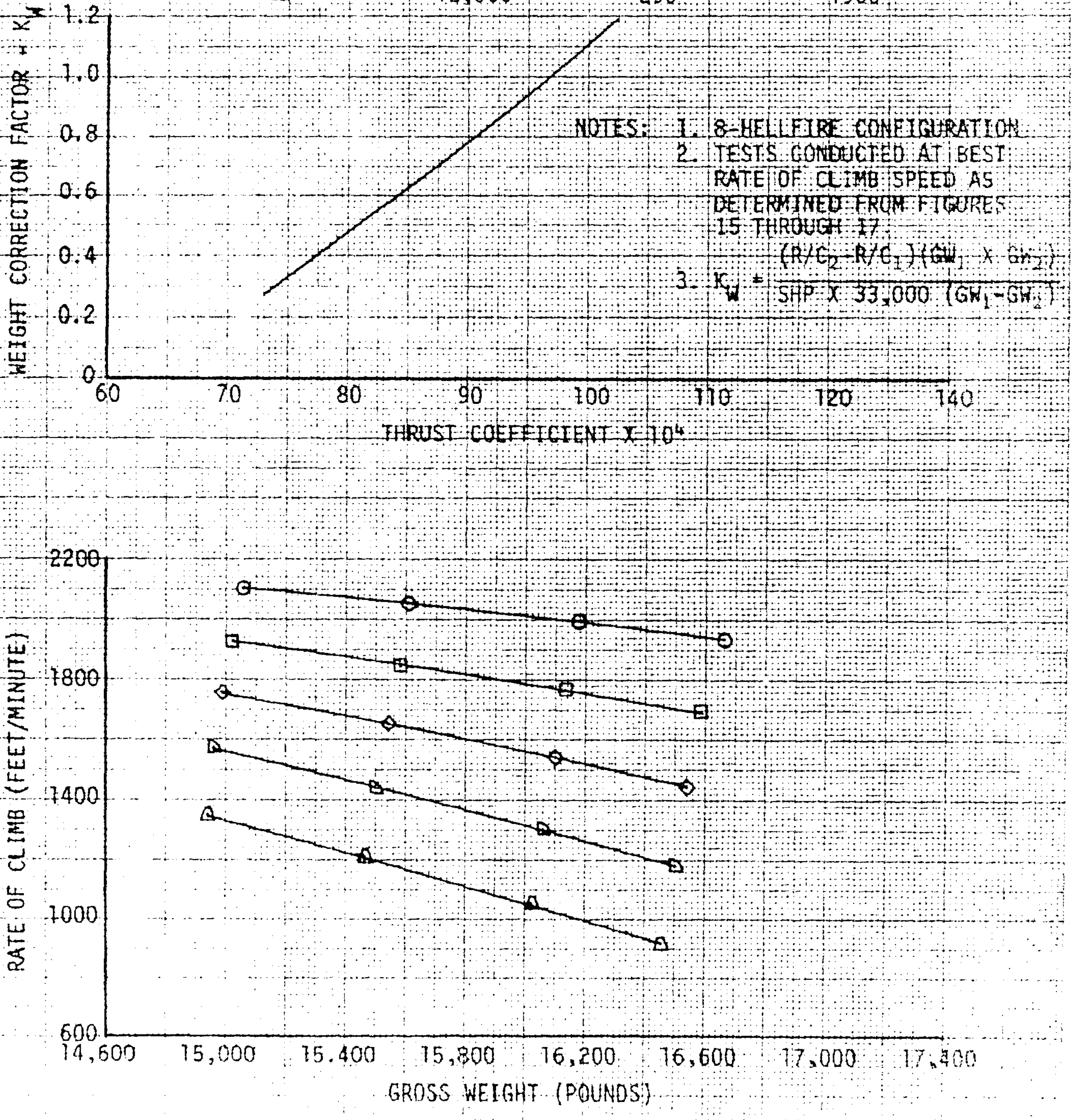
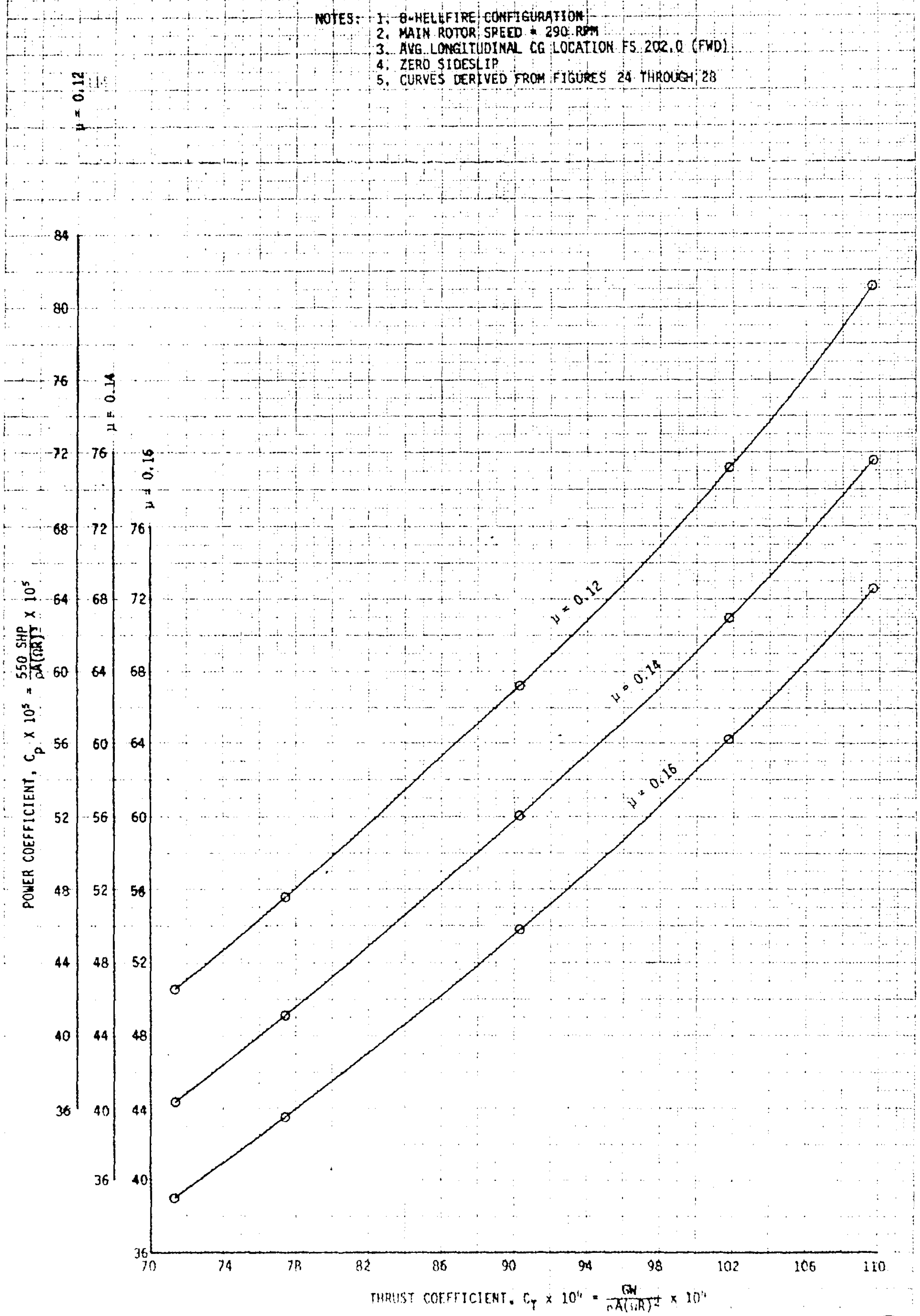


FIGURE 15
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258



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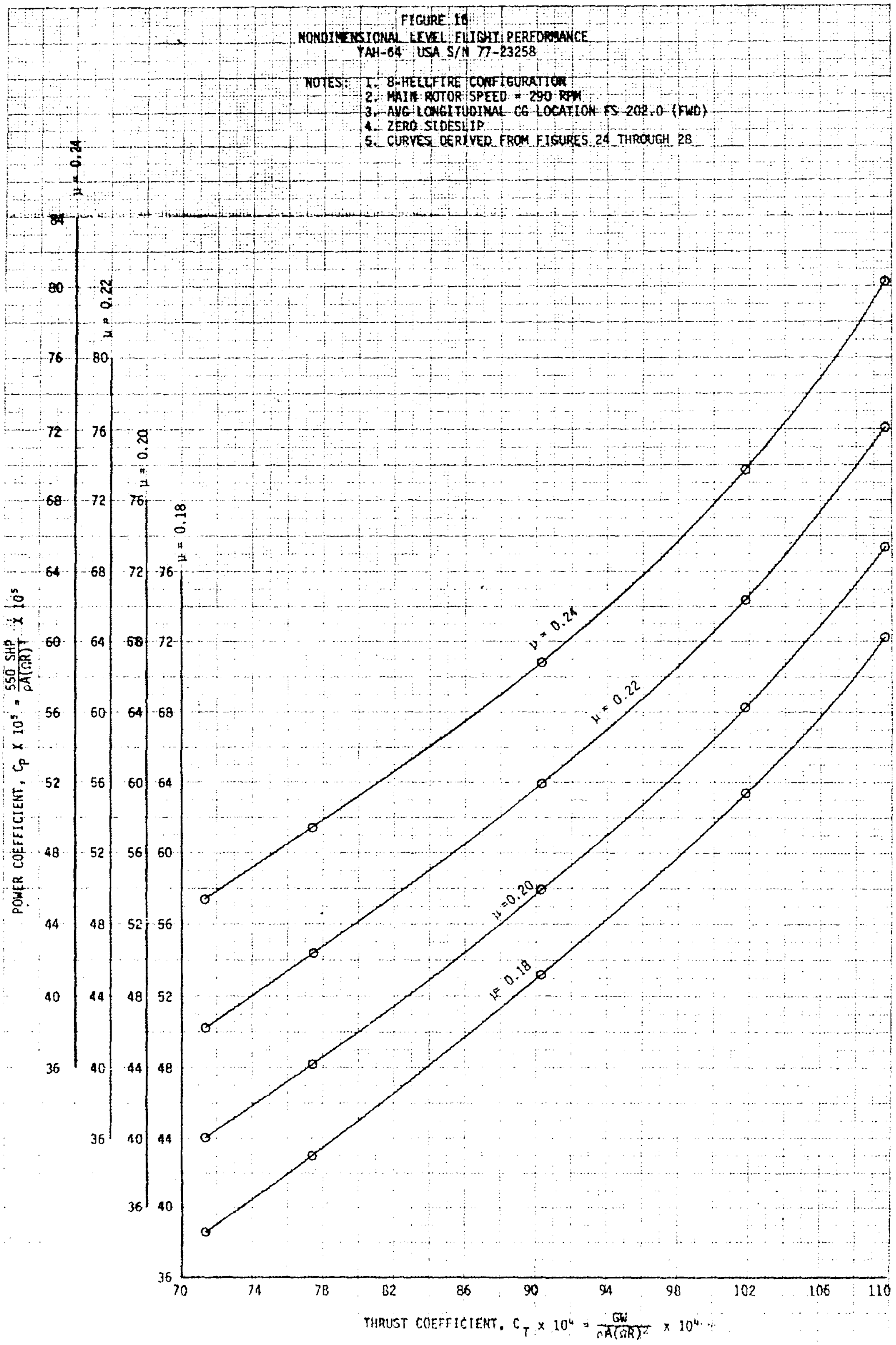

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FIGURE 17
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 24 THROUGH 28

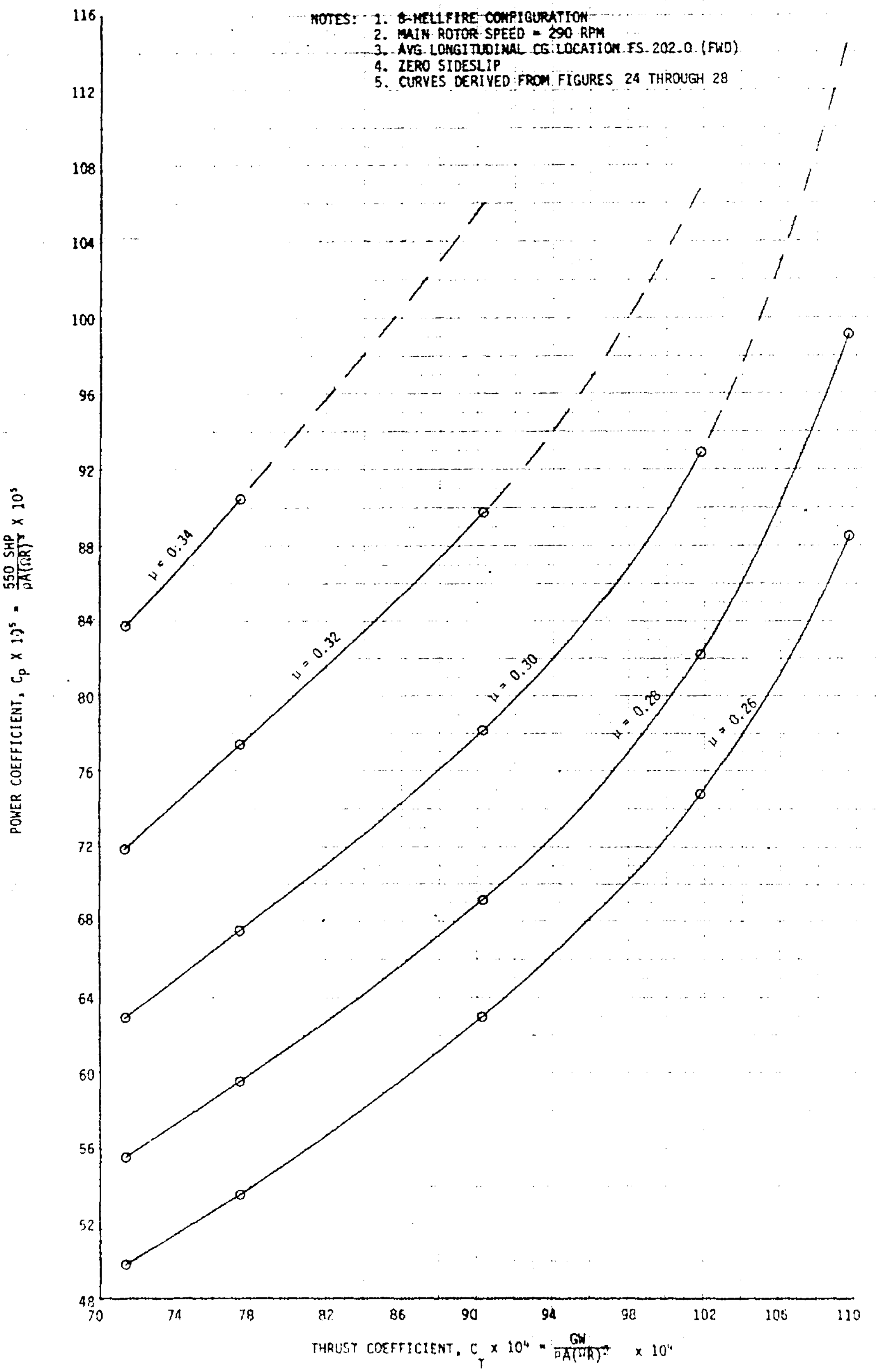

FIGURE 18
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES: 1. 16-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 201.8 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 29 THROUGH 32

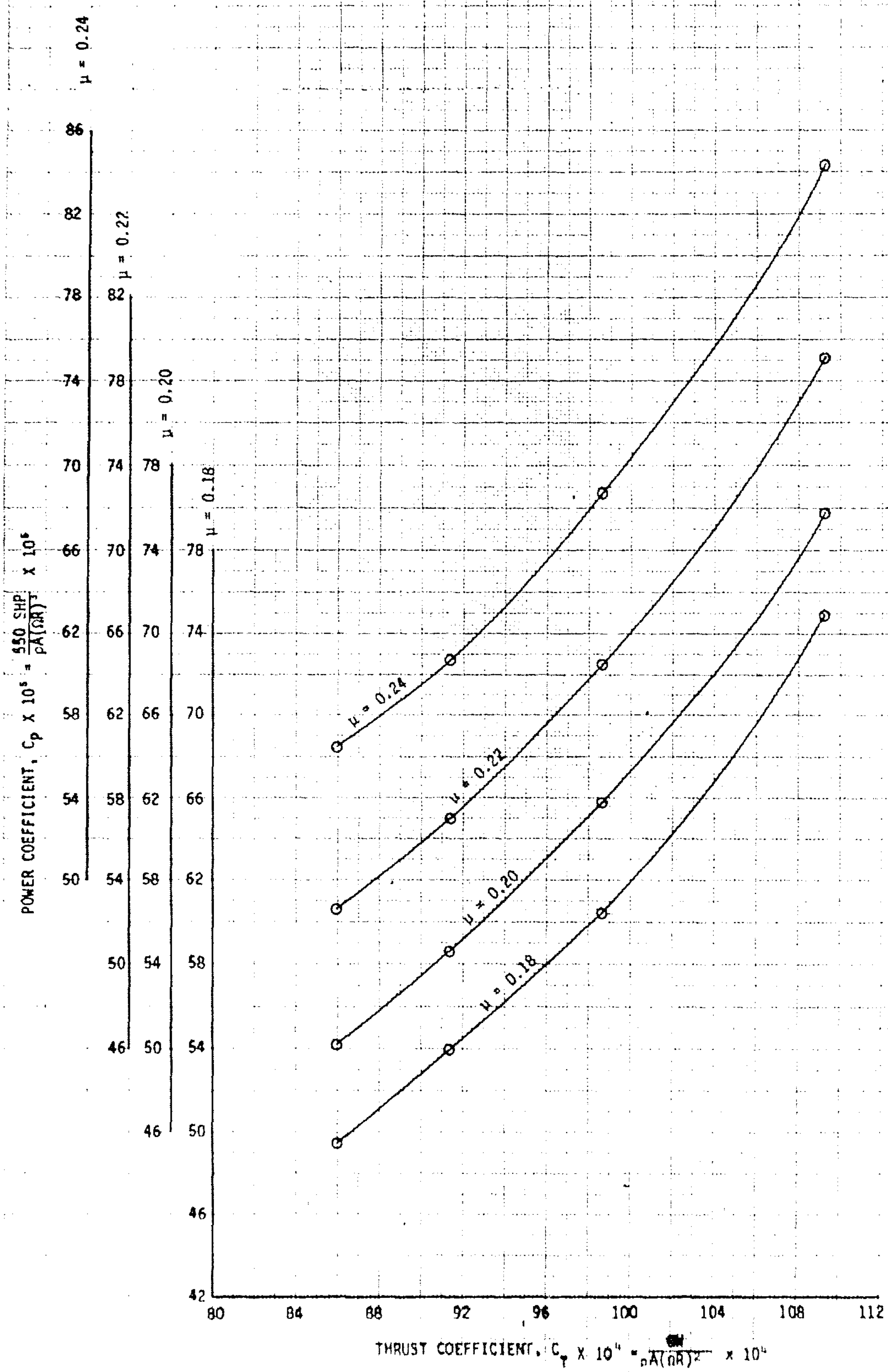

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FIGURE 19
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES:
1. 16-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 201.8 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 29 THROUGH 32

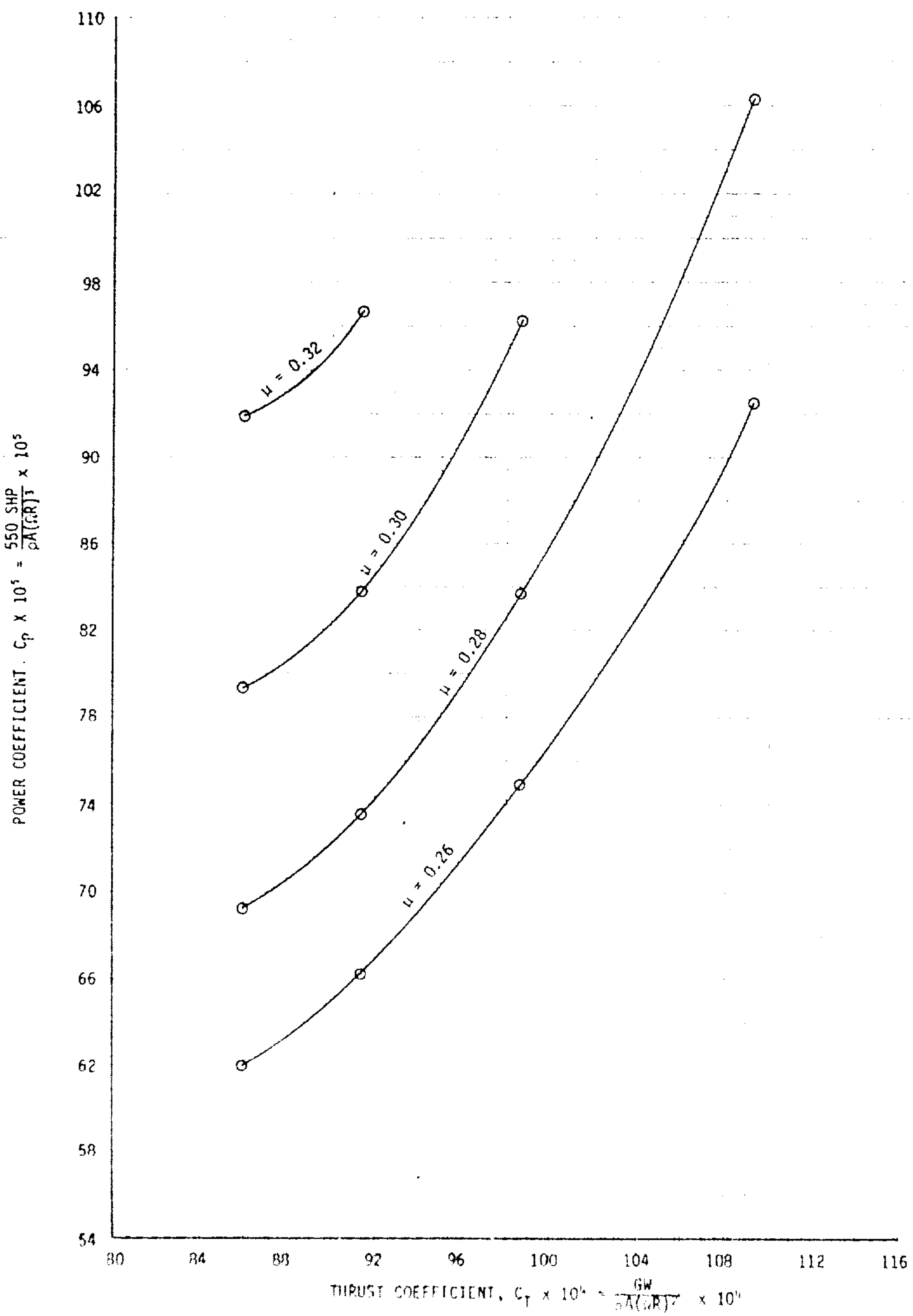

FIGURE 20
LEVEL FLIGHT RANGE PERFORMANCE
YAH-64 USA S/N 77-23258

CONFIGURATION: 8-HELLFIRE

NOTES:
1. DATA DERIVED FROM FIGURES 6 AND 7, APP D, AND 15 THROUGH 17, APP E.
2. SOLID LINE DENOTES 4000 FT, 35° C CONDITIONS
3. BROKEN LINE DENOTES SEA LEVEL STANDARD CONDITIONS

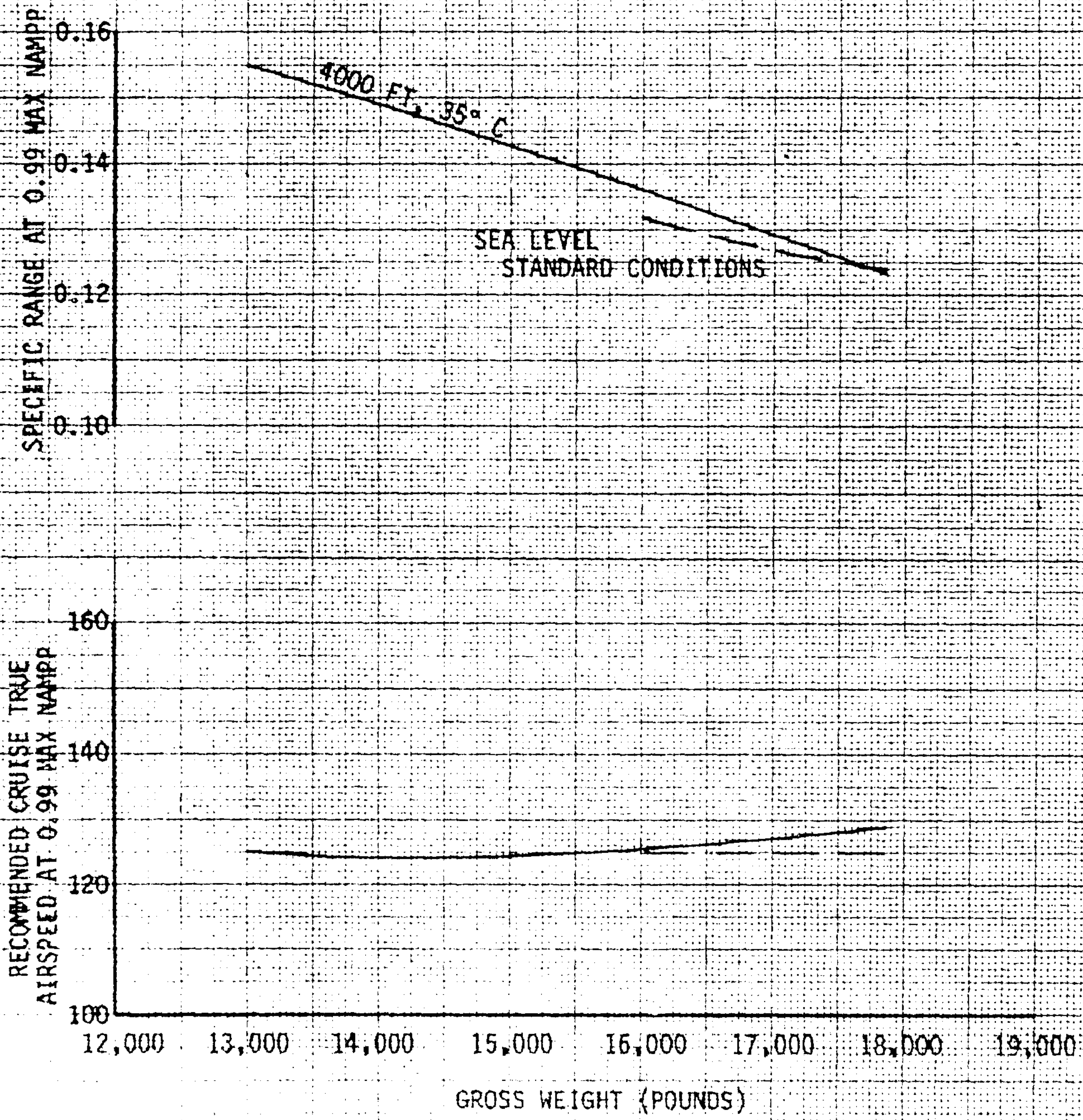

FIGURE C

LEVEL FLIGHT ENDURANCE SUMMARY

YAH-64 USA S/N 77-23258

CONFIGURATION: 8-HELLFIRE

NOTES: 1. DATA DERIVED FROM FIGURES 6 AND 7, APP C, AND 15 THROUGH 17, APP E.
2. SOLID LINE DENOTES 4000 FT, 35° C CONDITIONS
3. BROKEN LINE DENOTES SEA LEVEL STANDARD CONDITIONS

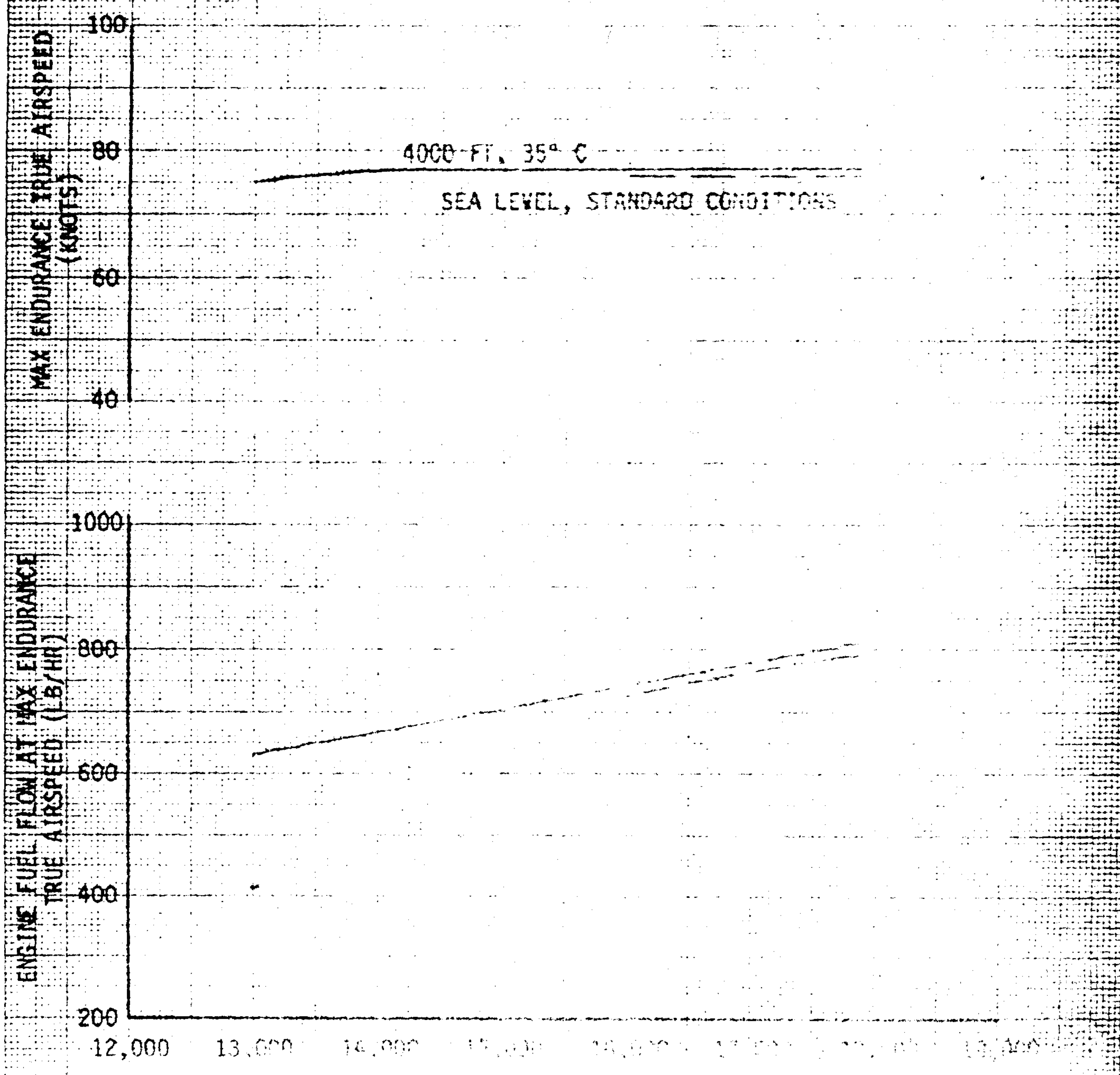

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FIGURE 22
CHANGE IN EQUIVALENT FLAT PLATE AREA WITH SIDESLIP
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG ADVANCE RATIO	AVG $C_T \times 10^4$
○	15,780	202.8(FWD)	0.8(LT)	8620	10.0	290	0.35	89.56
□	15,300	202.7(FWD)	0.8(LT)	8960	17.5	290	0.33	87.76
◇	15,040	202.7(FWD)	0.8(LT)	9980	6.0	290	0.32	89.07
▽	15,360	202.6(FWD)	0.8(LT)	9460	8.0	290	0.27	89.99
▽	15,000	202.7(FWD)	0.8(LT)	10,420	7.0	290	0.22	90.07

NOTES:
1. 8-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE BALL-CENTERED DATA
3. $C_{P_{ZERO\ SIDESLIP}}$ OBTAINED FROM FIGURES 15 THROUGH 17

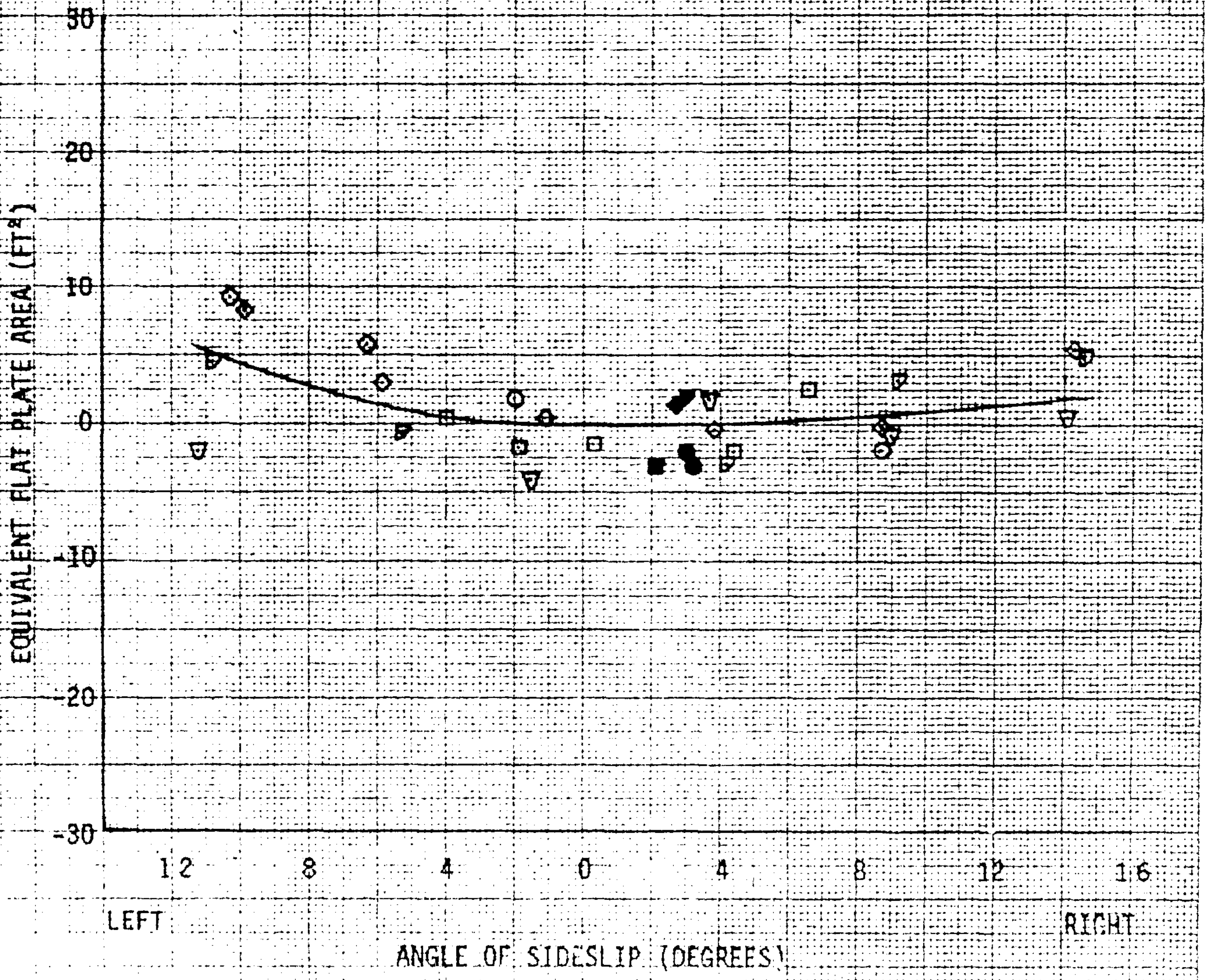
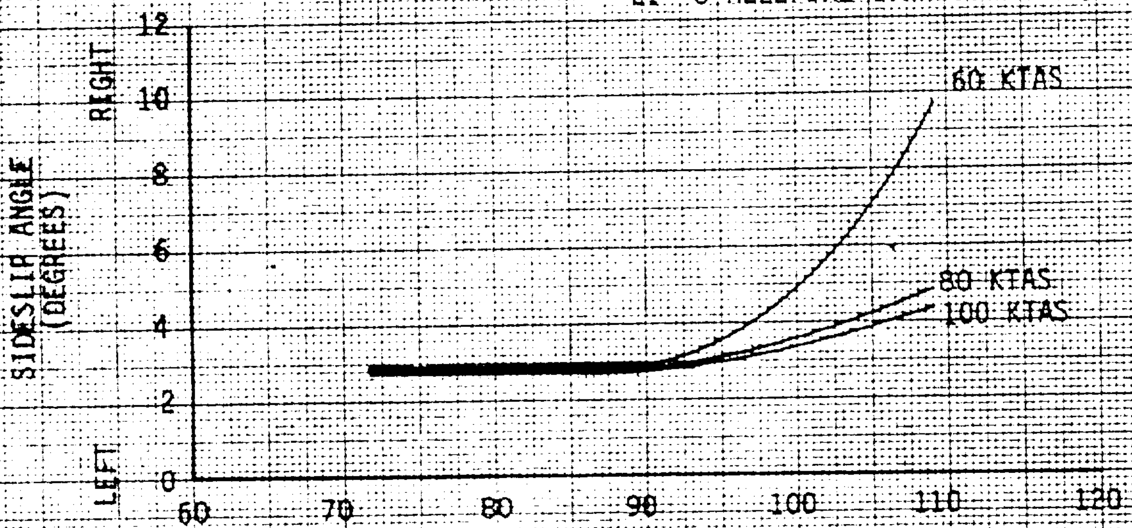


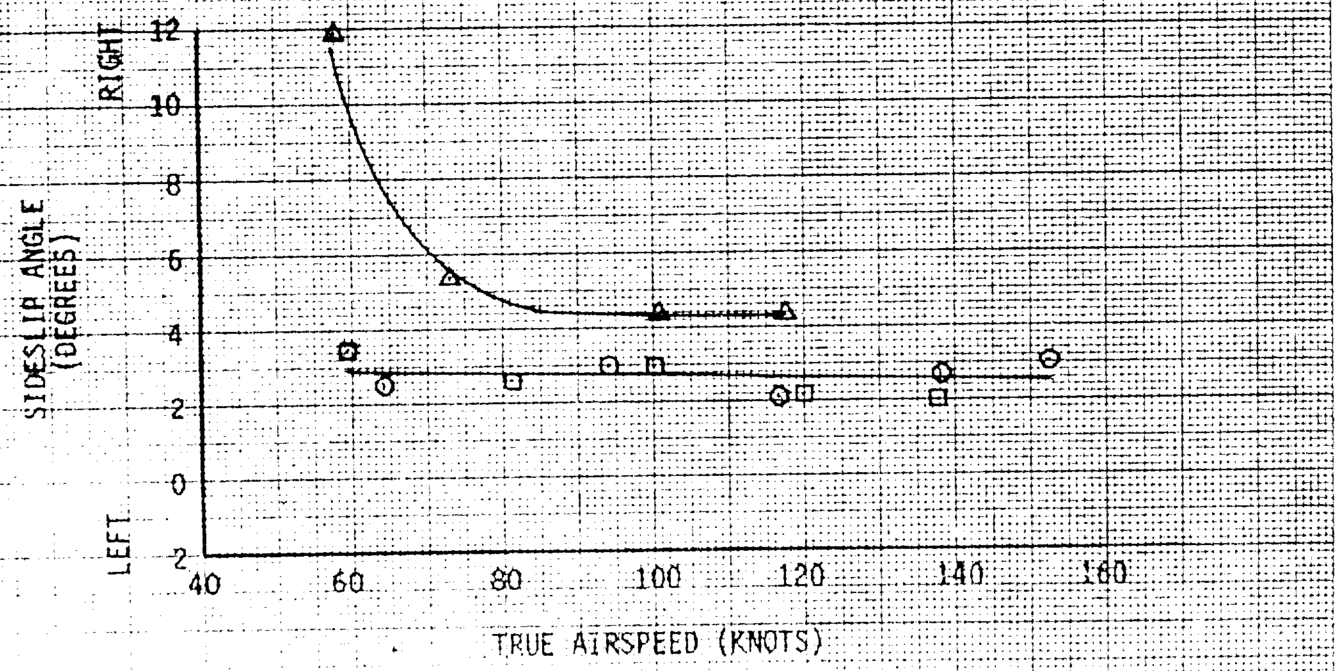
FIGURE 23
INHERENT SIDESLIP ANGLE
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T x 10^4
○	14,940	202.0(FWD)	0.8(LT)	3140	29.0	290	71.75
□	15,460	202.0(FWD)	0.8(LT)	9240	8.0	290	89.46
△	15,360	202.1(FWD)	0.8(LT)	15,640	0.5	290	109.06

NOTES: 1. DATA OBTAINED IN COORDINATED BALL-CENTERED FLIGHT.
2. 8-HELLFIRE CONFIGURATION, DASE ON



THRUST COEFFICIENT, $C_T \times 10^4 = \frac{GW}{\rho A(\Omega R)^2} \times 10^4$



TRUE AIRSPEED (KNOTS)

FIGURE 24
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $X10^4$	CONFIGURATION
14,820	202.0(FWD)	0.8(LT)	3240	25.5	290	71.39	8-HELLFIRE

NOTE: ZERO SIDESLIP

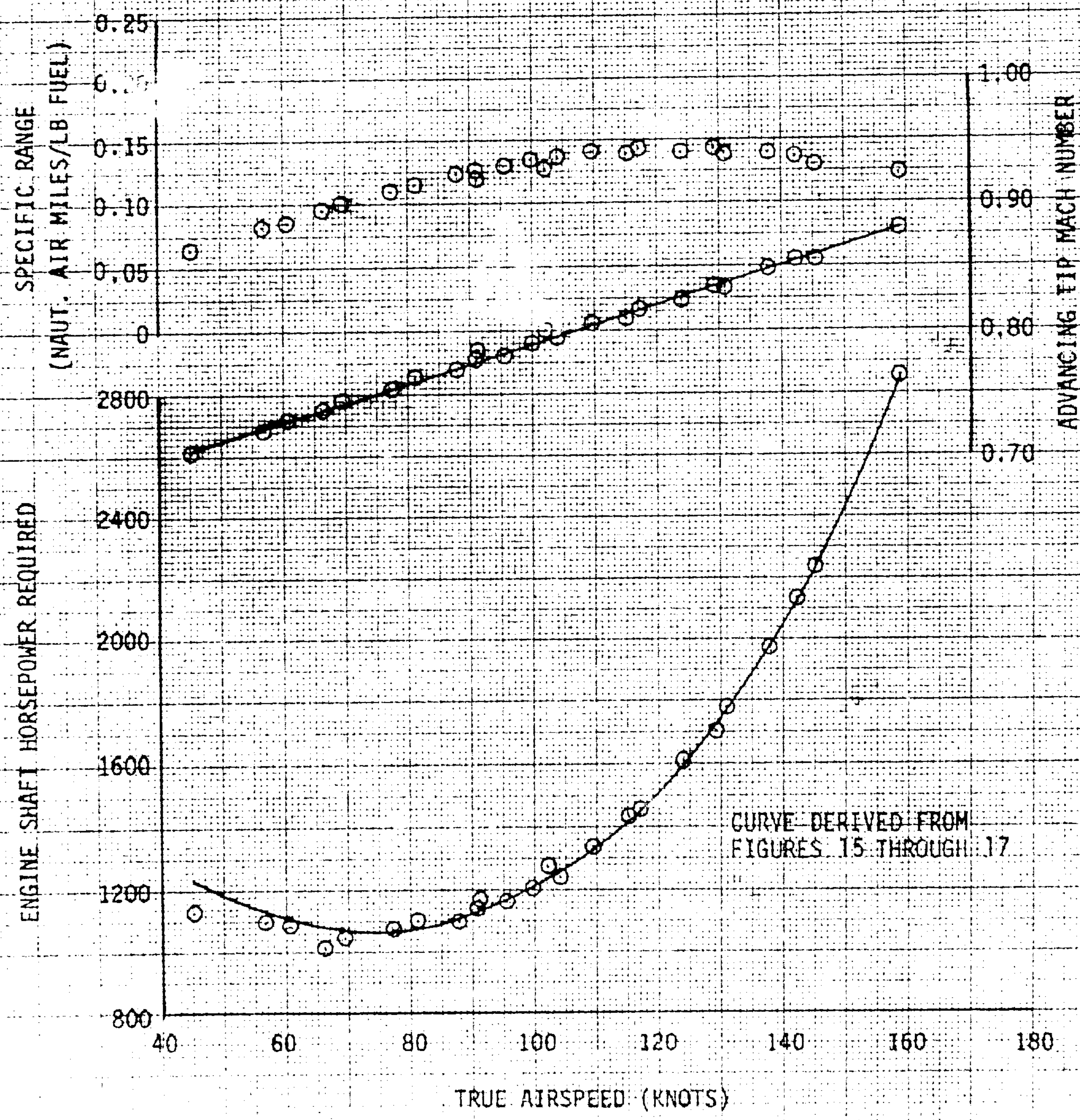


FIGURE 25
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $x10^4$	CONFIGURATION
14,960	202.0(FWD)	0.0(MID)	5620	29.5	290	77.43	8-HELLFIRE

NOTE: ZERO SIDESLIP

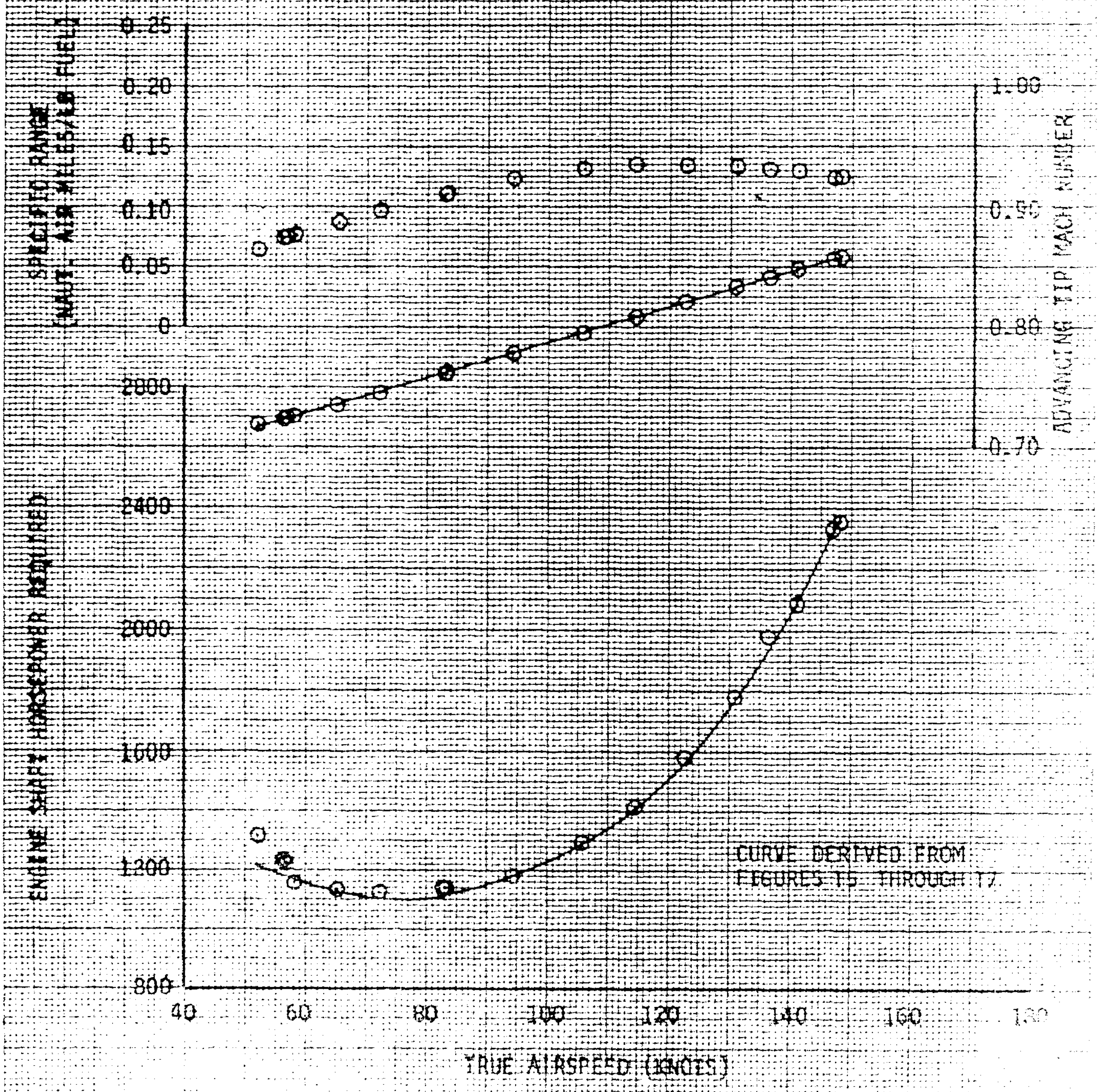


FIGURE 26
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $C_T \times 10^4$	CONFIGURATION
15,500	202.2(FWD)	0.0(MID)	9460	16.0	290	90.31	8 HELLFIRE

NOTE: ZERO SIDESLIP

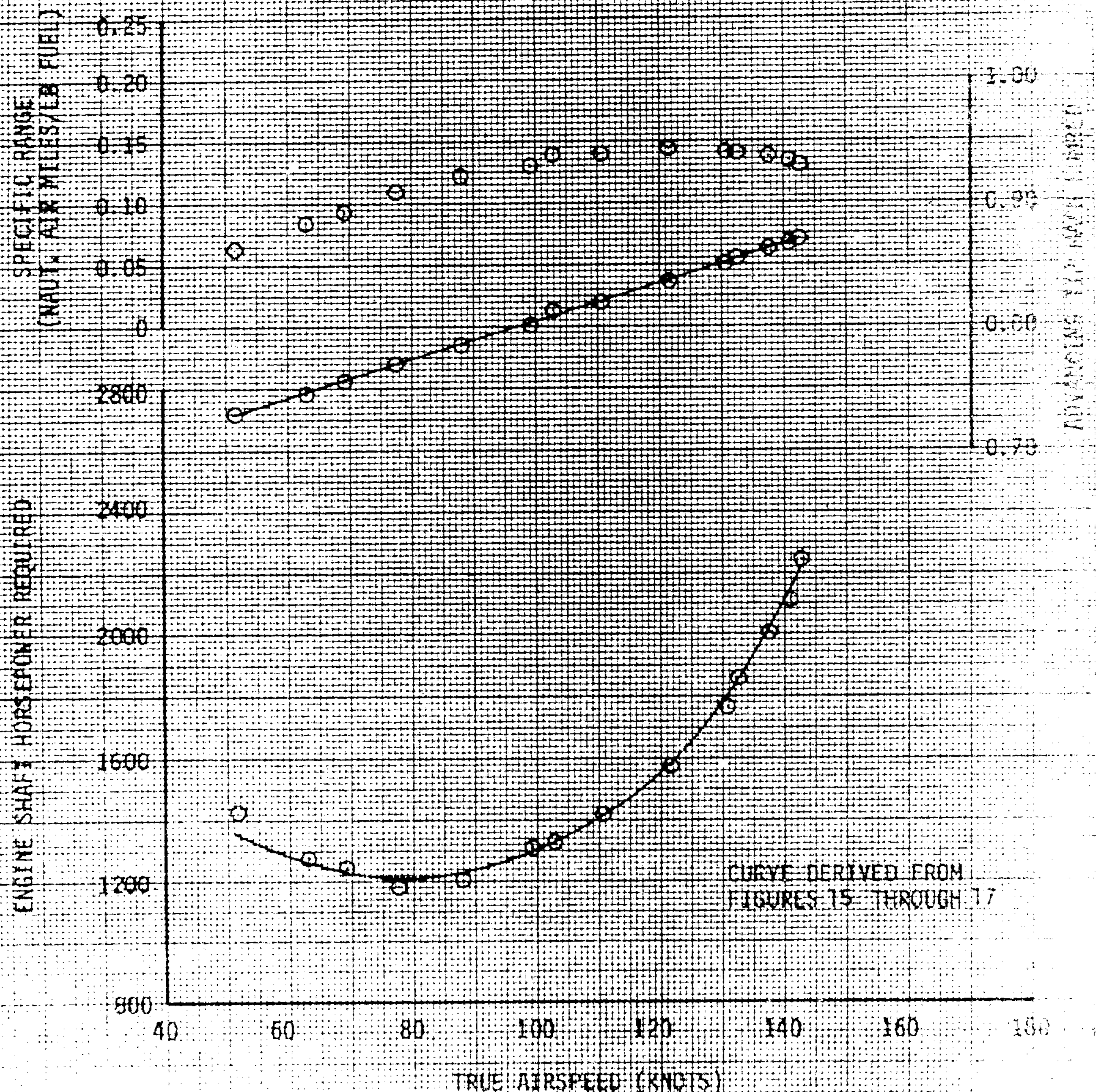


FIGURE 27
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $C_T \times 10^4$	CONFIGURATION
15,400	201.9(FWD)	0.0(MID)	13,220	5.5	289	101.78	8-HELLFIRE

NOTE: ZERO SIDESLIP

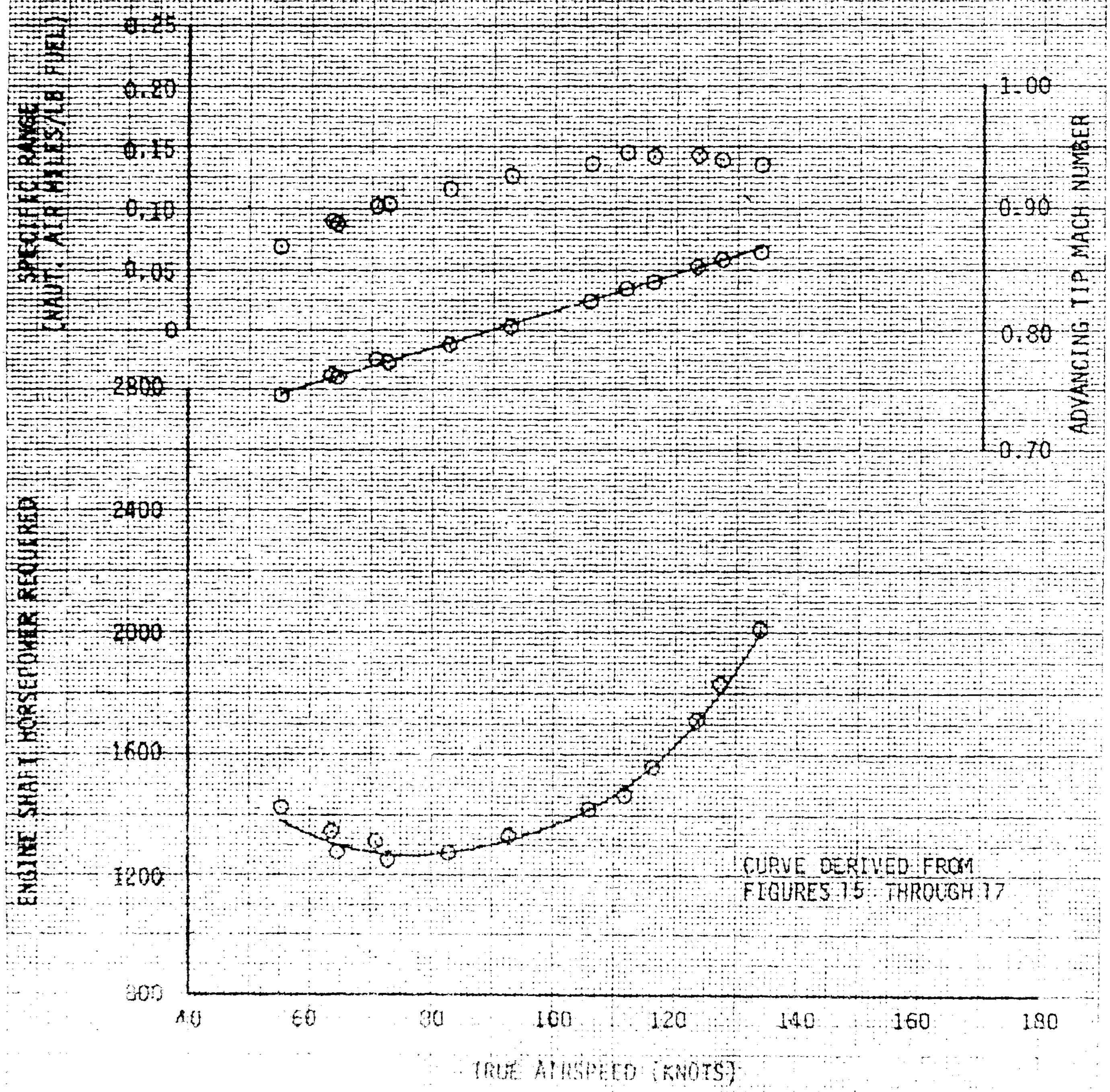


FIGURE 28
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T X10^4	CONFIGURATION
15,320	202.0(FWD)	0.8(LT)	15,880	2.5	290	109.63	8-HELLFIRE

NOTE: ZERO SIDESLIP

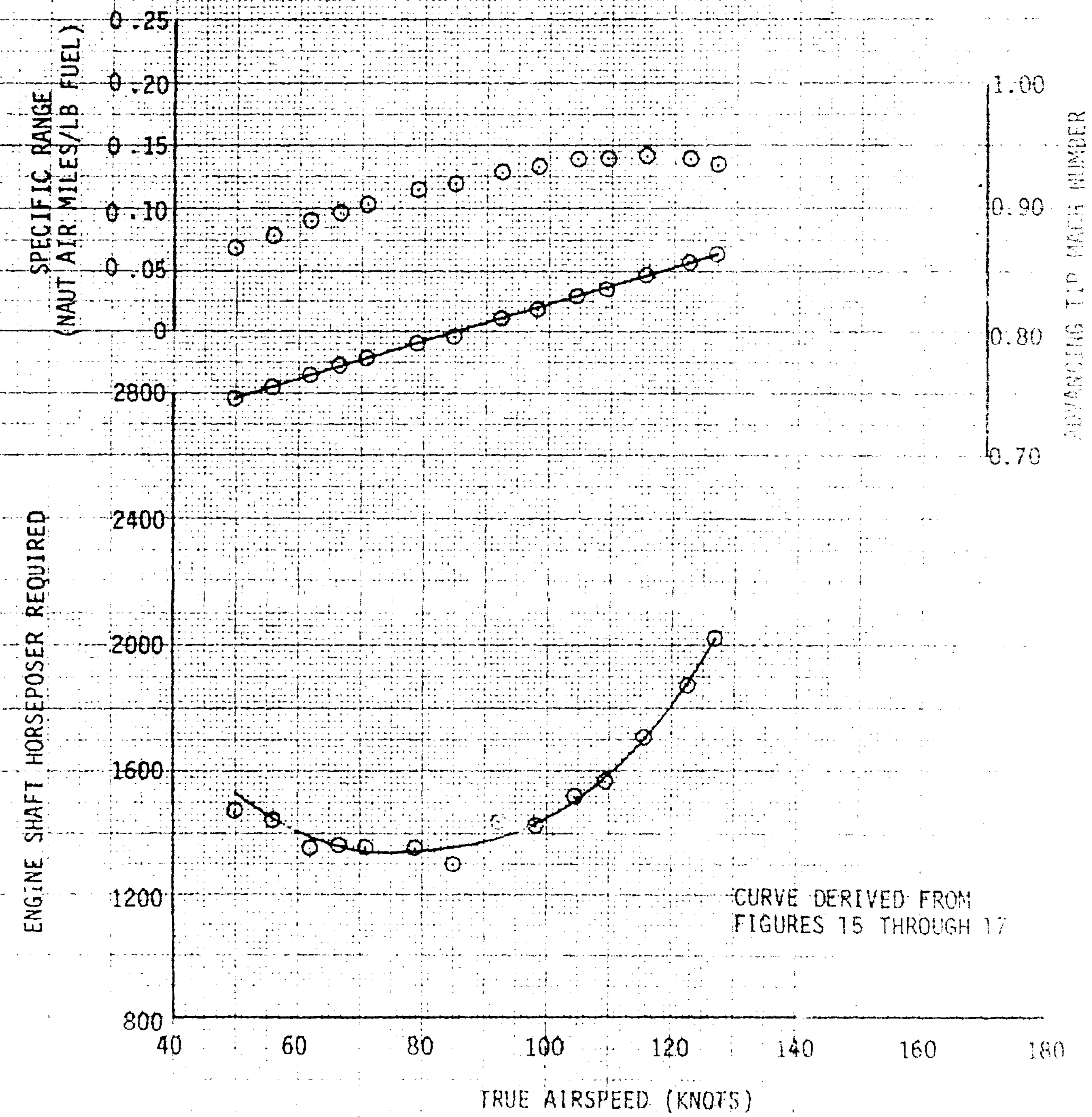


FIGURE 29
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $C_T \times 10^4$	CONFIGURATION
15,780	201.3(FWD)	0.0(MID)	7300	20.5	290	85.98	16-HELLFIRE

NOTE: ZERO SIDESLIP

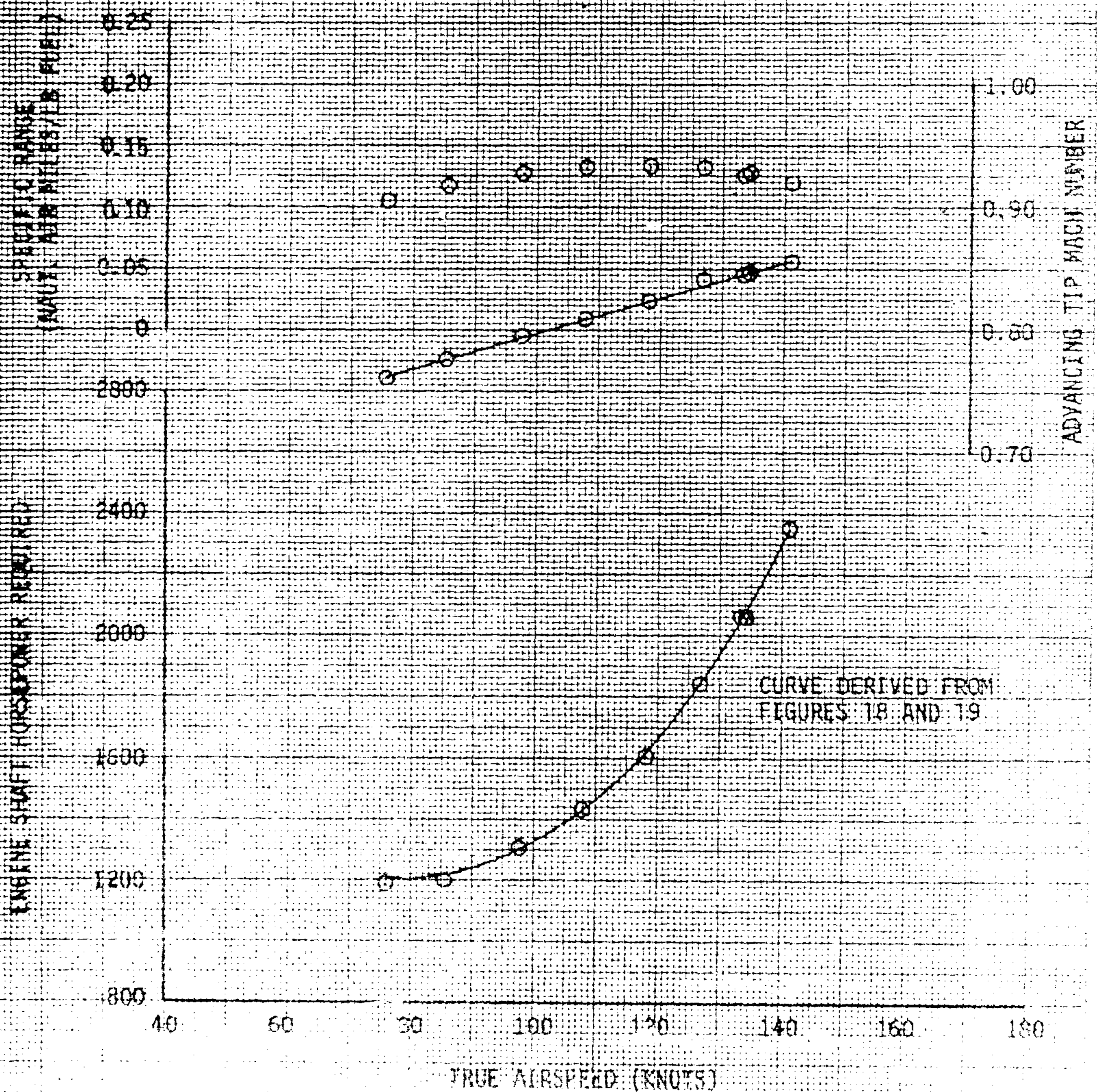


FIGURE 30
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $\times 10^4$	CONFIGURATION
16,620	202.2(FWD)	0.8(LT)	7360	23.0	289	91.35	16-HELLFIRE

NOTE: ZERO SIDESLIP

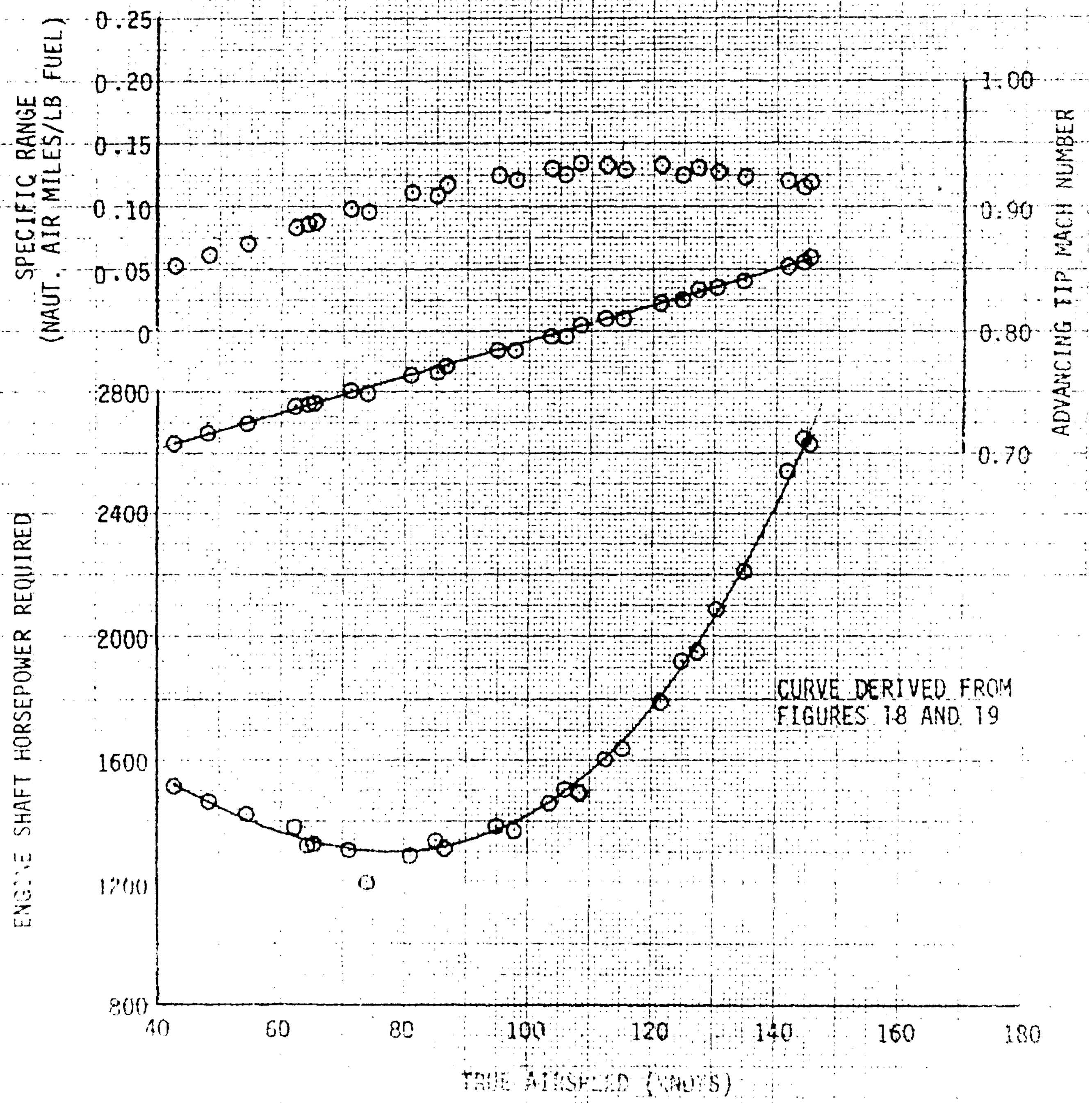


FIGURE 37
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $C_T \times 10^4$	CONFIGURATION
16,420	201.4 (FWD)	0.0 (MID)	10,440	11.0	290	98.66	16-HELLFIRE

NOTE: ZERO SIDESLIP

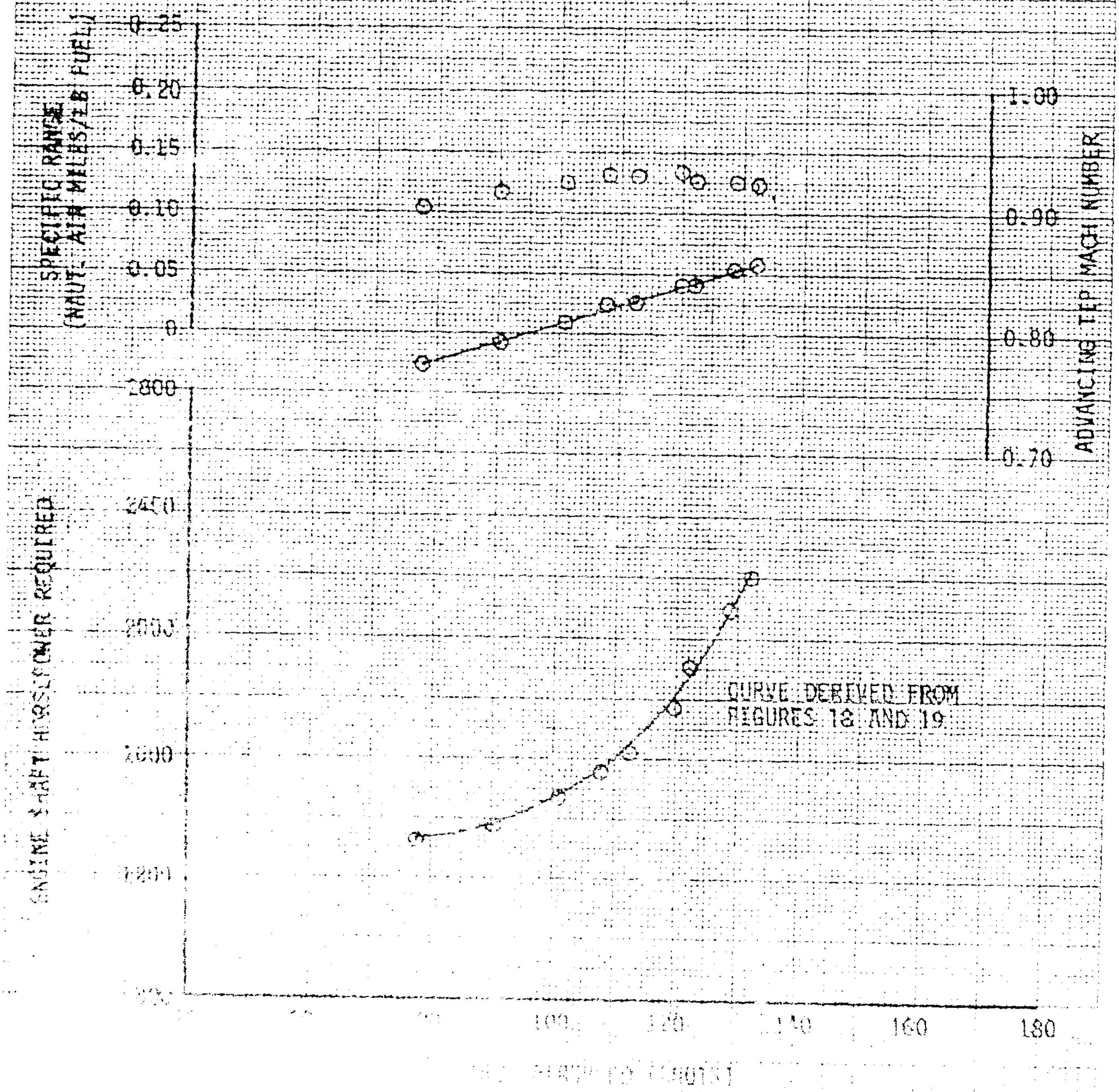


FIGURE 32
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T x104	CONFIGURATION
16,620	202.2(FWD)	0.8(LT)	13,260	3.5	290	109.23	16-HELLFIRE

NOTE: ZERO SIDESLIP

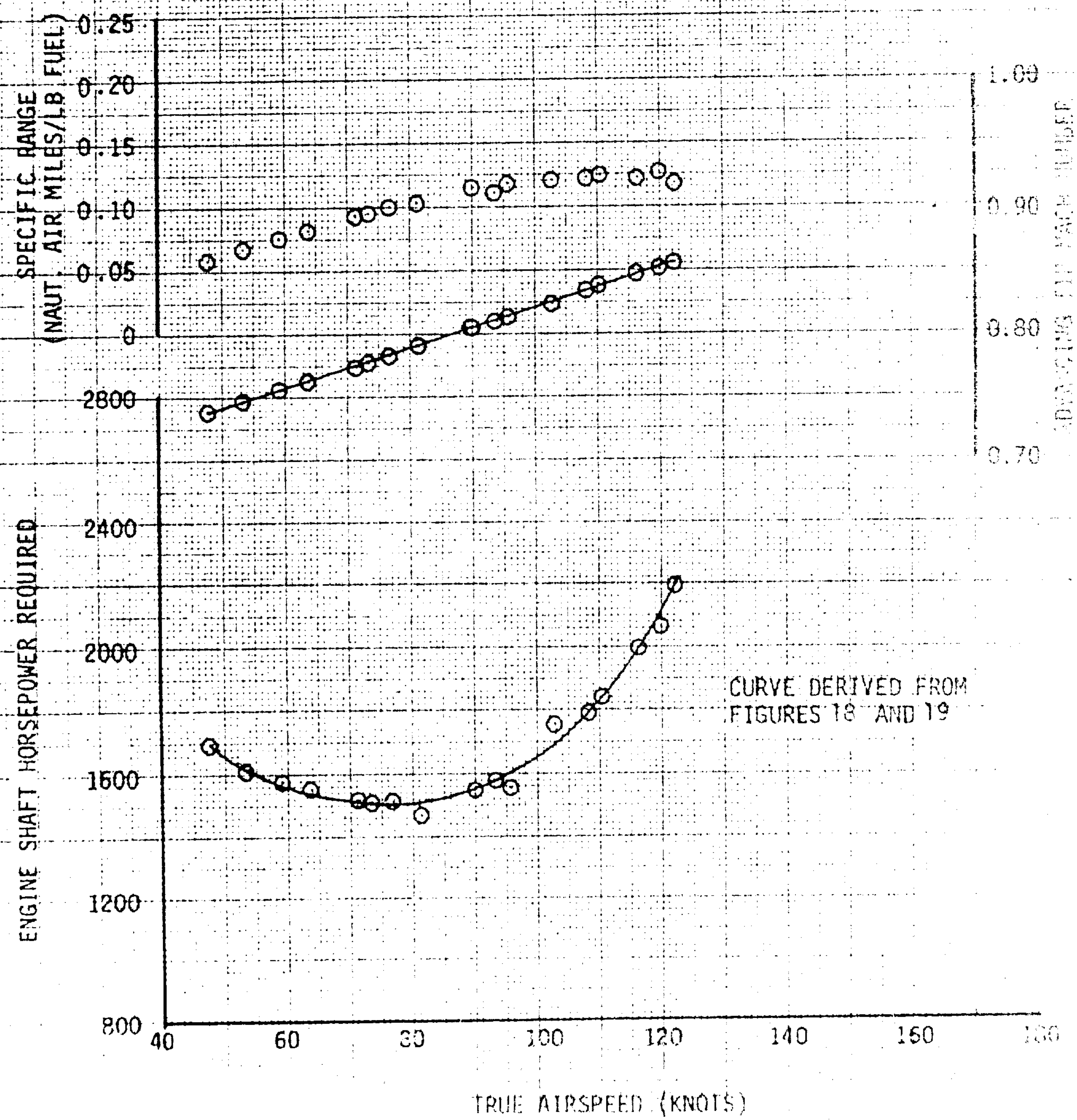


FIGURE 33
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $\times 10^4$	CONFIGURATION
14,280	204.1(FWD)	0.8(LT)	16,260	-1.5	289	104.19	CLEAN

NOTE: ZERO SIDESLIP

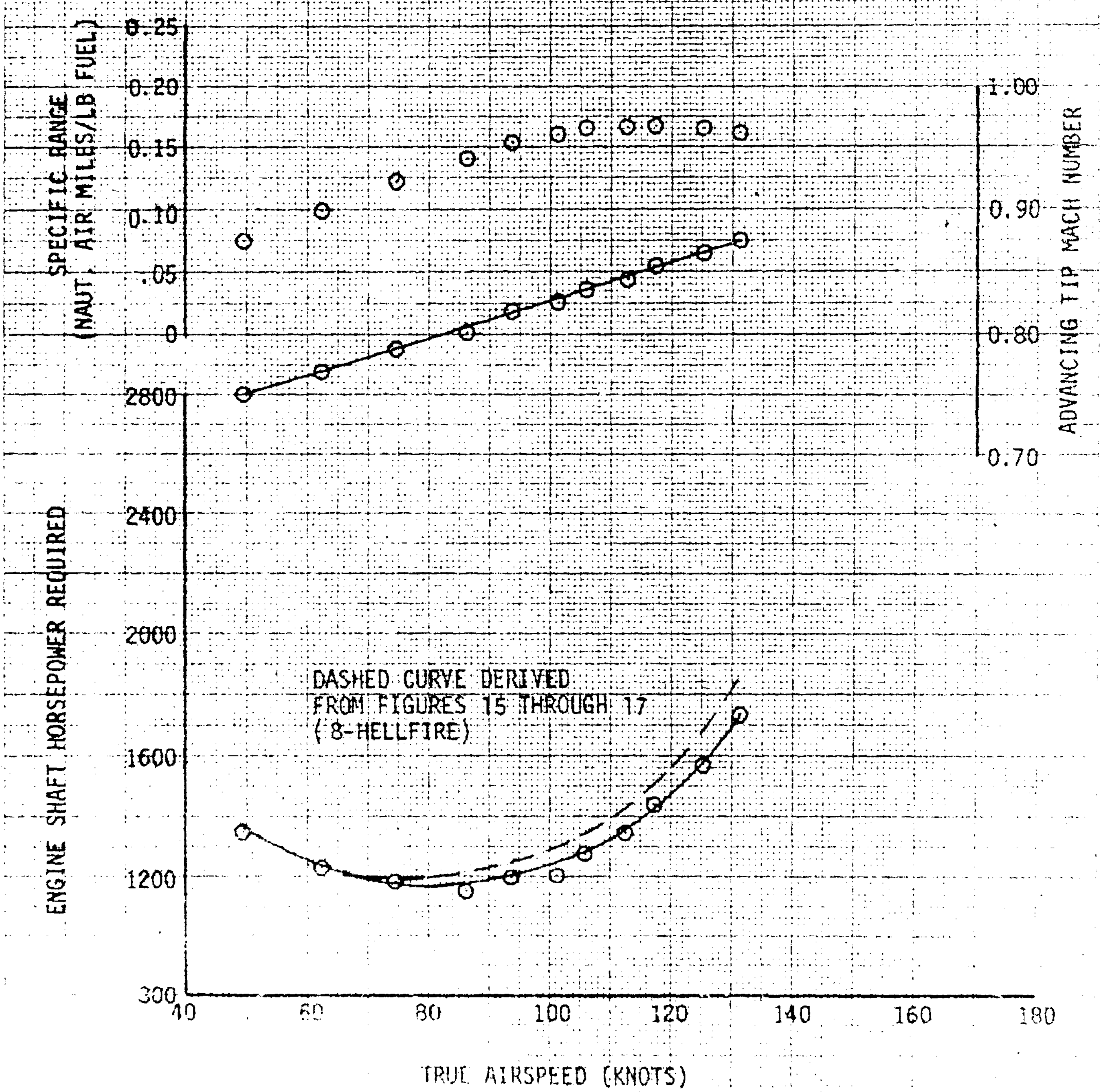


FIGURE 34
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $\times 10^4$	CONFIGURATION
14,200	203.9(FWD)	0.8(LT)	11,640	13.0	289	69.61	CLEAN

NOTE: ZERO SIDESLIP

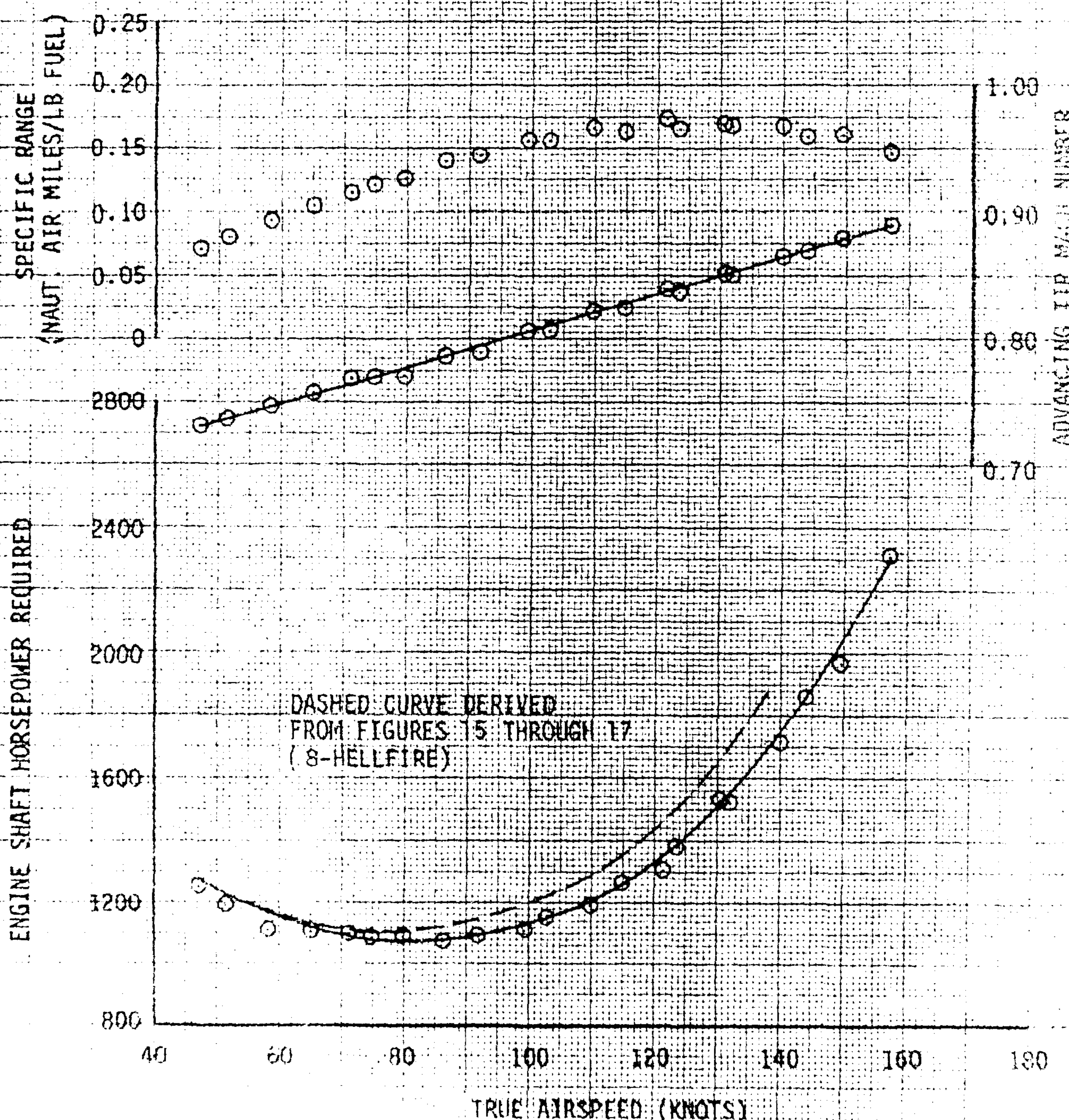


FIGURE 35
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $\times 10^4$	CONFIGURATION
14,080	203.9(FWD)	0.8(LT)	6220	29.0	289	74.73	CLEAN

NOTE: ZERO SIDESLIP

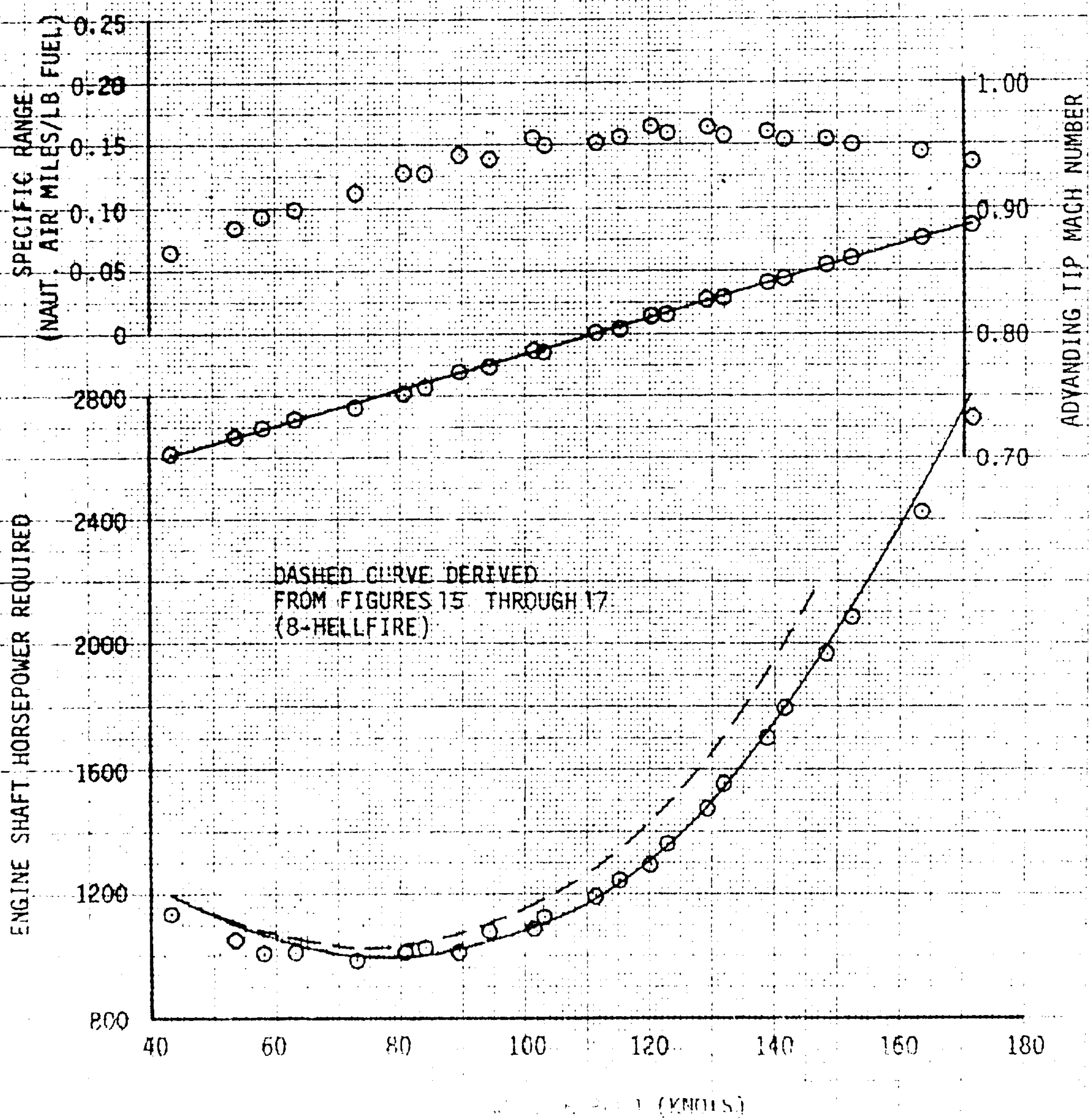


FIGURE 36
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $\times 10^4$	CONFIGURATION
13,760	206.9(AFT)	0.8(LT)	12,760	8.5	289	89.61	CLEAN

NOTE: ZERO SIDESLIP

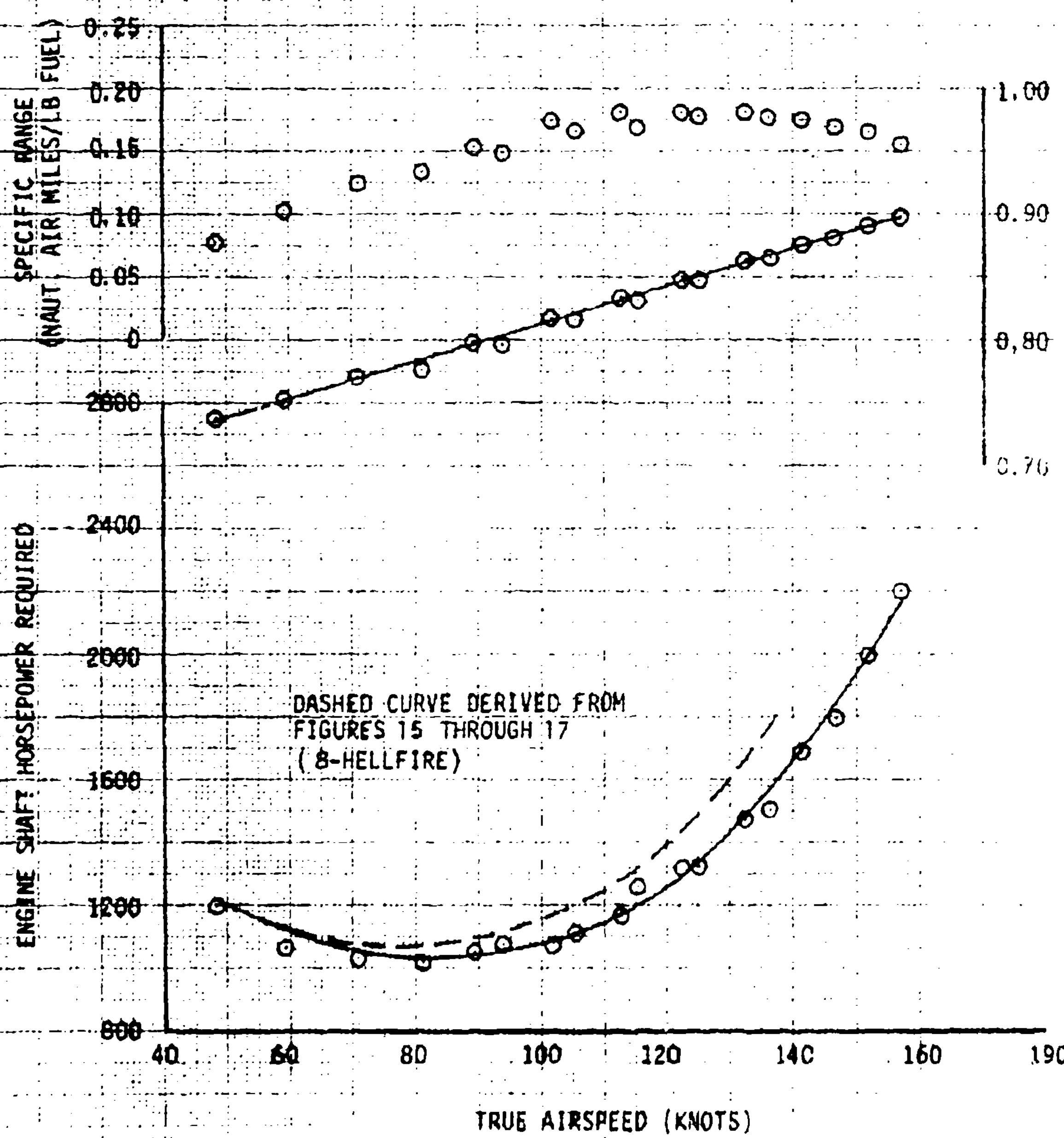


FIGURE 37
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $\times 10^4$	CONFIGURATION
13,960	206.8(AFT)	0.8(LT)	6680	25.5	290	74.63	CLEAN

NOTE: ZERO SIDESLIP

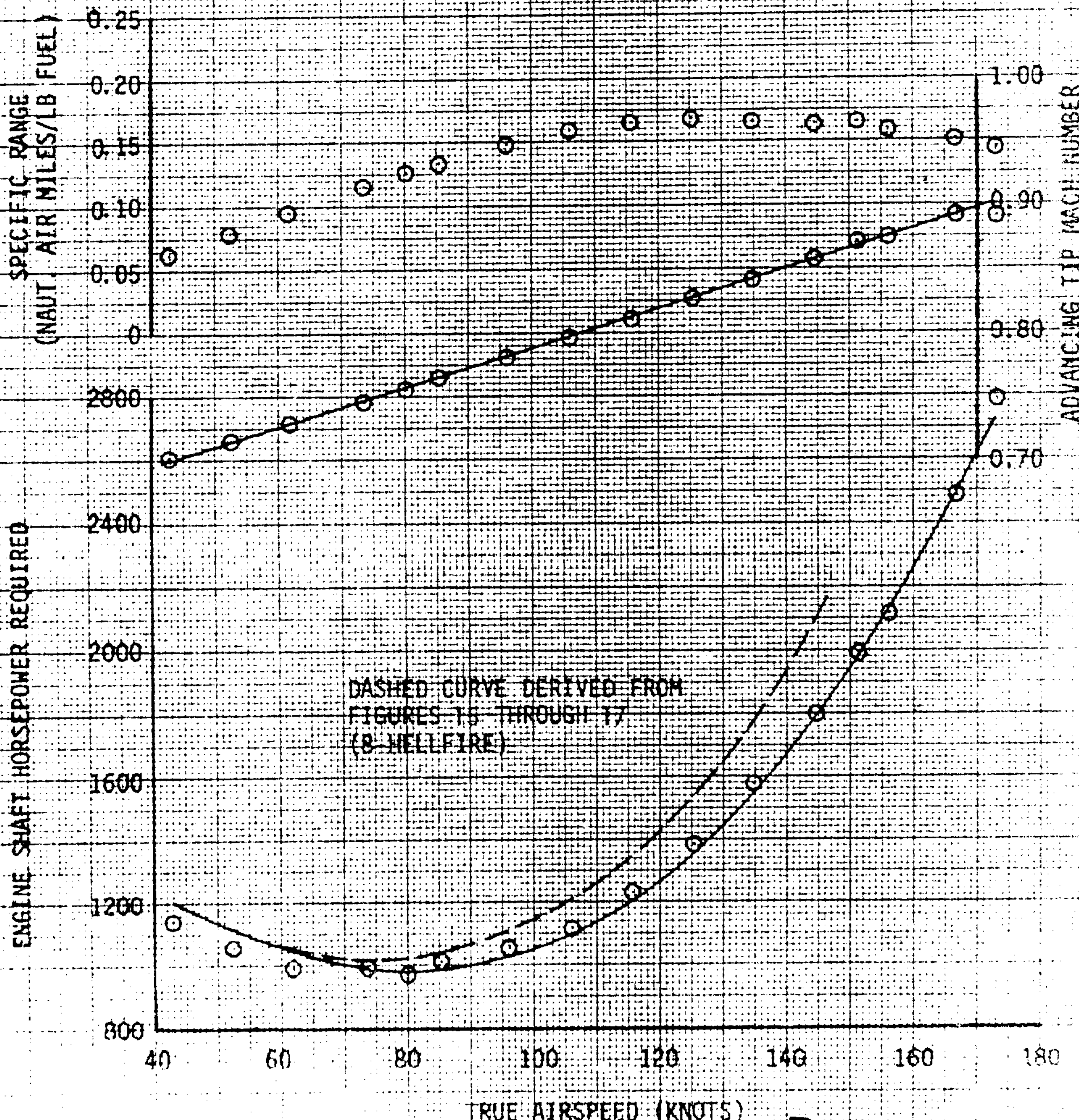


FIGURE 38
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $X10^4$	CONFIGURATION
14,640	202.2(FWD)	0.8(LT)	11,120	13.5	290	89.87	2-FERRY TANK

NOTE: ZERO SIDESLIP

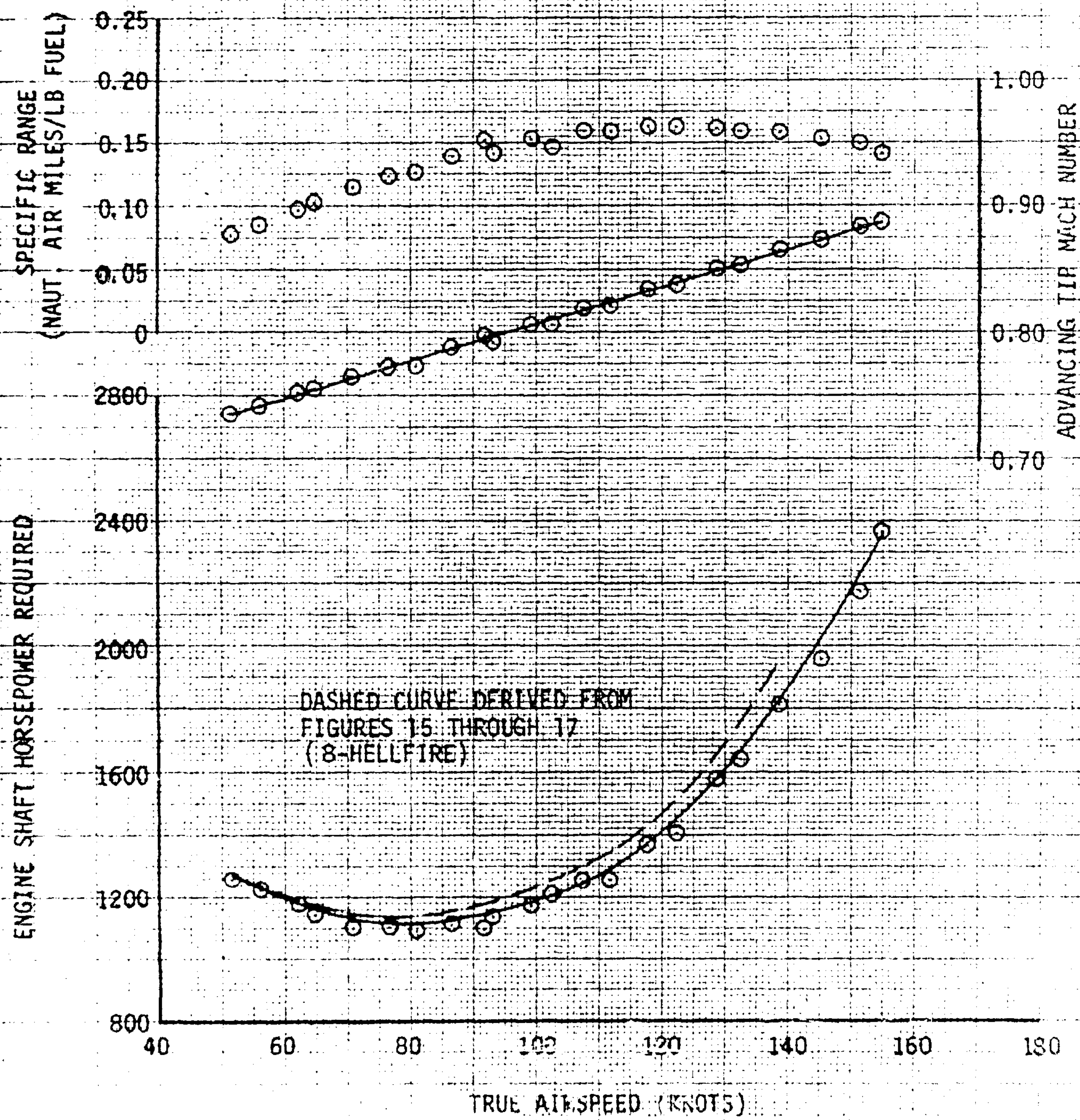


FIGURE 39
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T $x10^4$	CONFIGURATION
14,680	202.2(FWD)	0.9(LT)	15,560	0.5	291	103.24	2-FERRY TANK

NOTE: ZERO SIDESLIP

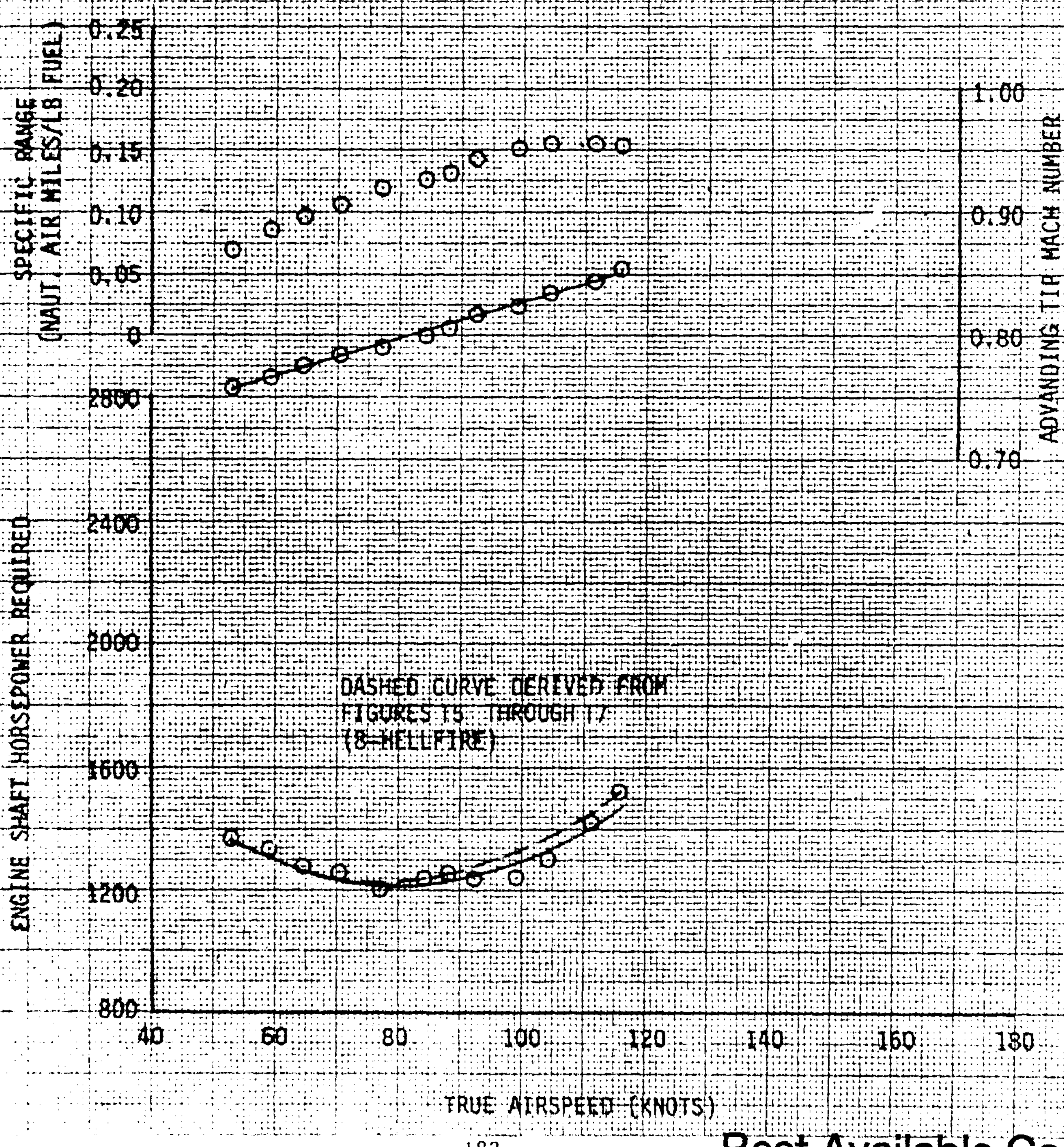


FIGURE 40
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T X 10^4
14,694	202.0 (FWD)	0.0	7120	35.0	289	80.18

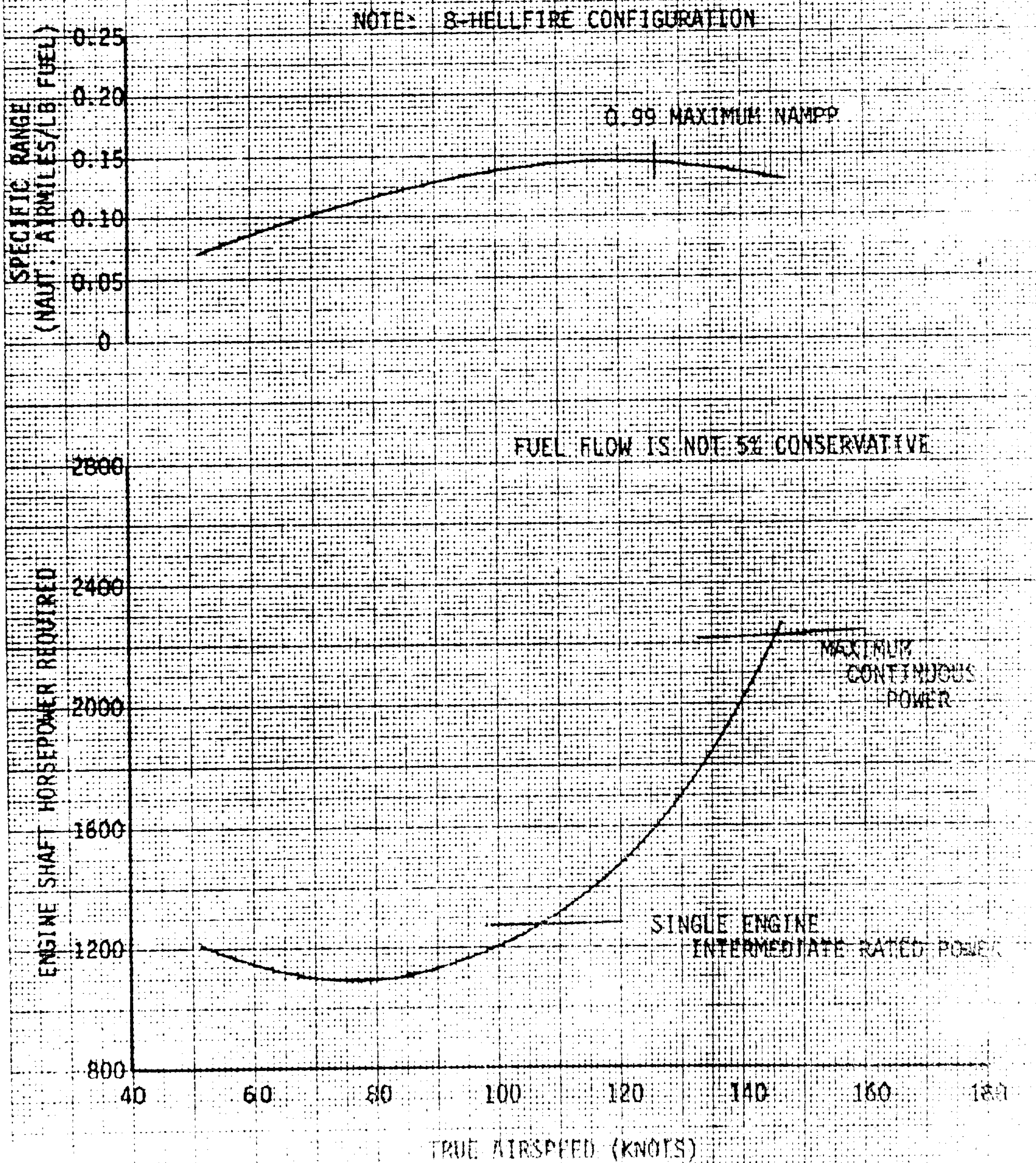


FIGURE 41
AUTOROTATIONAL DESCENT PERFORMANCE
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
⊙	15,200	205.3(MID)	0.8(LT)	4960	20.0	289
⊡	15,440	204.5(MID)	0.8(LT)	10,180	12.0	289

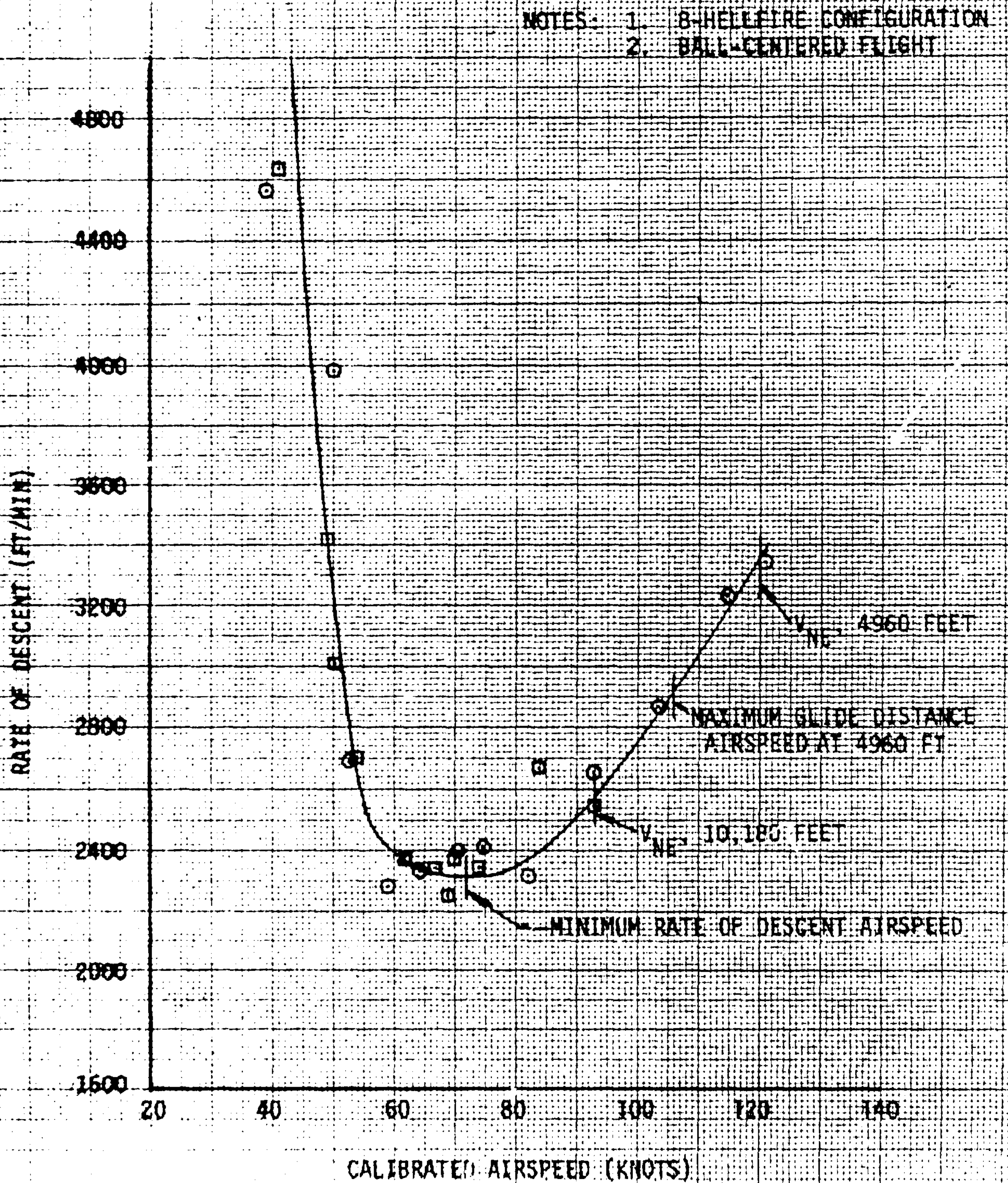


FIGURE 42
AUTOROTATIONAL DESCENT PERFORMANCE
YAH-64 USA S/N 77-23258

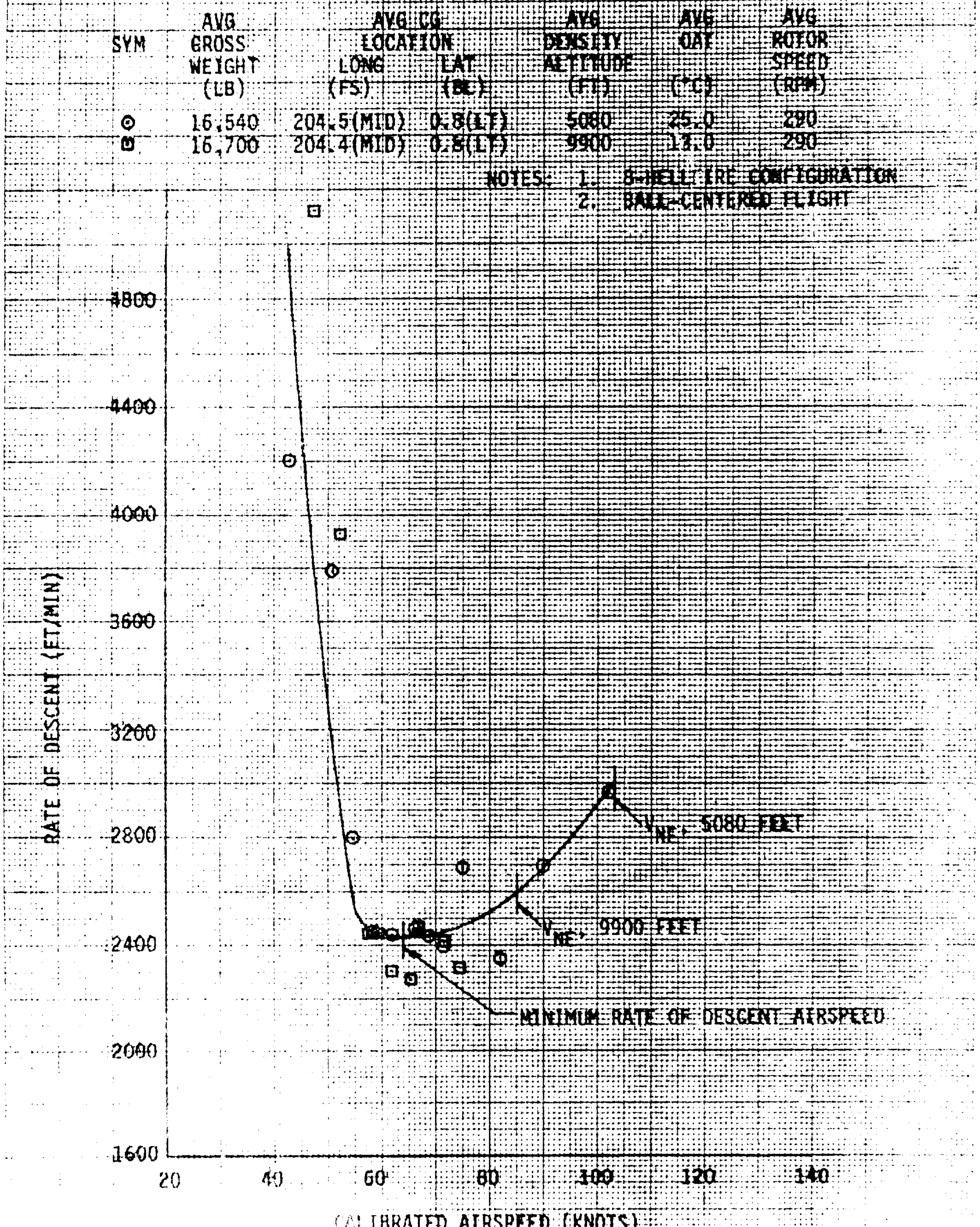


FIGURE 43
AUTOROTATIONAL DESCENT PERFORMANCE
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG CALIBRATED AIRSPEED (KTS)
⊙	14,860	205.3(MID)	0.8(LT)	5040	21.0	77
⊡	15,380	204.5(MID)	0.8(LT)	10,160	12.0	70

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT

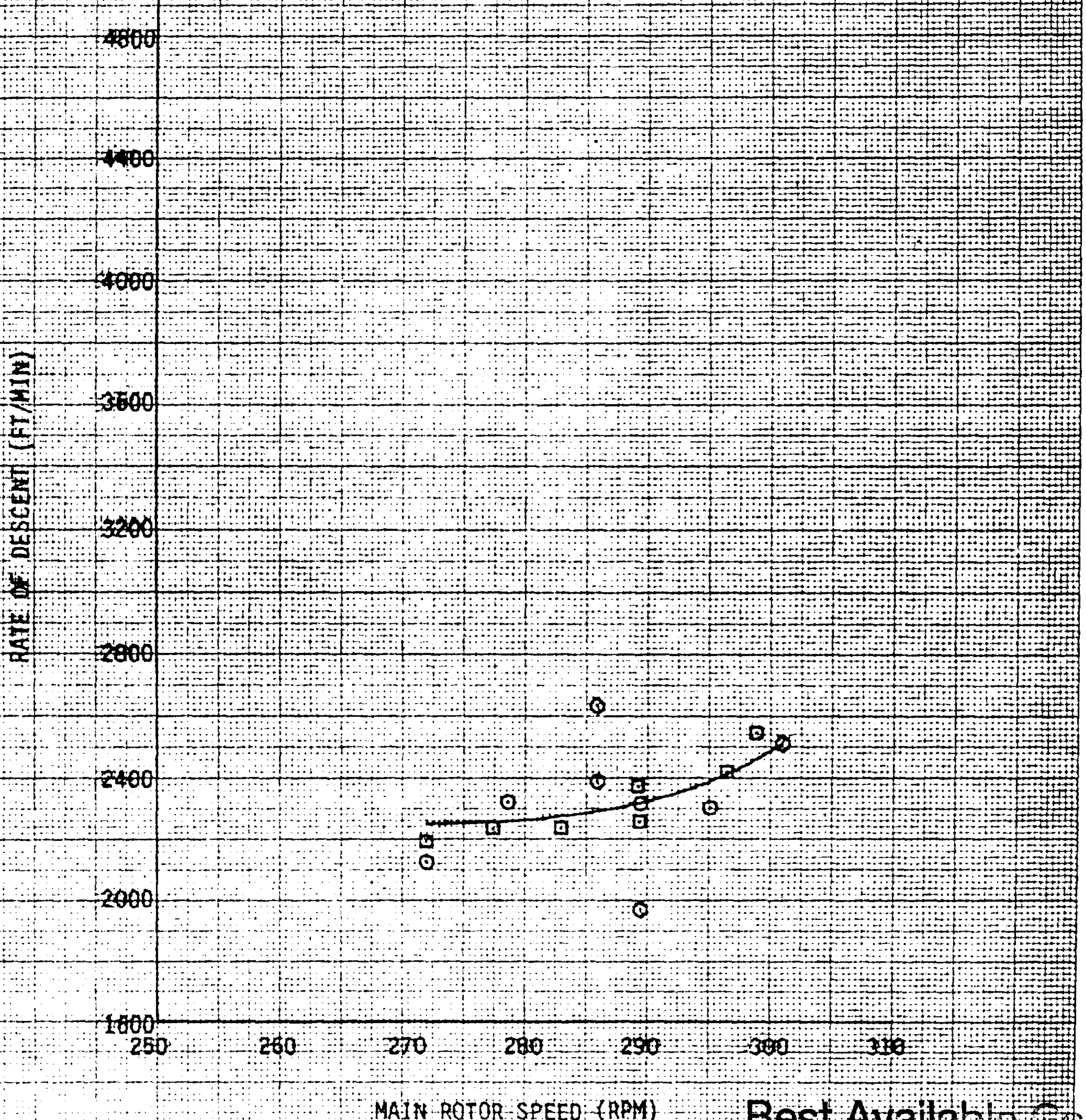


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FIGURE 44
AUTOROTATIONAL DESCENT PERFORMANCE
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG CALIBRATED AIRSPEED (KTS)
⊙	16,300	204.5(MID)	0.8(LT)	5080	25.0	68
⊡	16,420	204.3(MID)	0.8(LT)	9960	13.0	59

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT

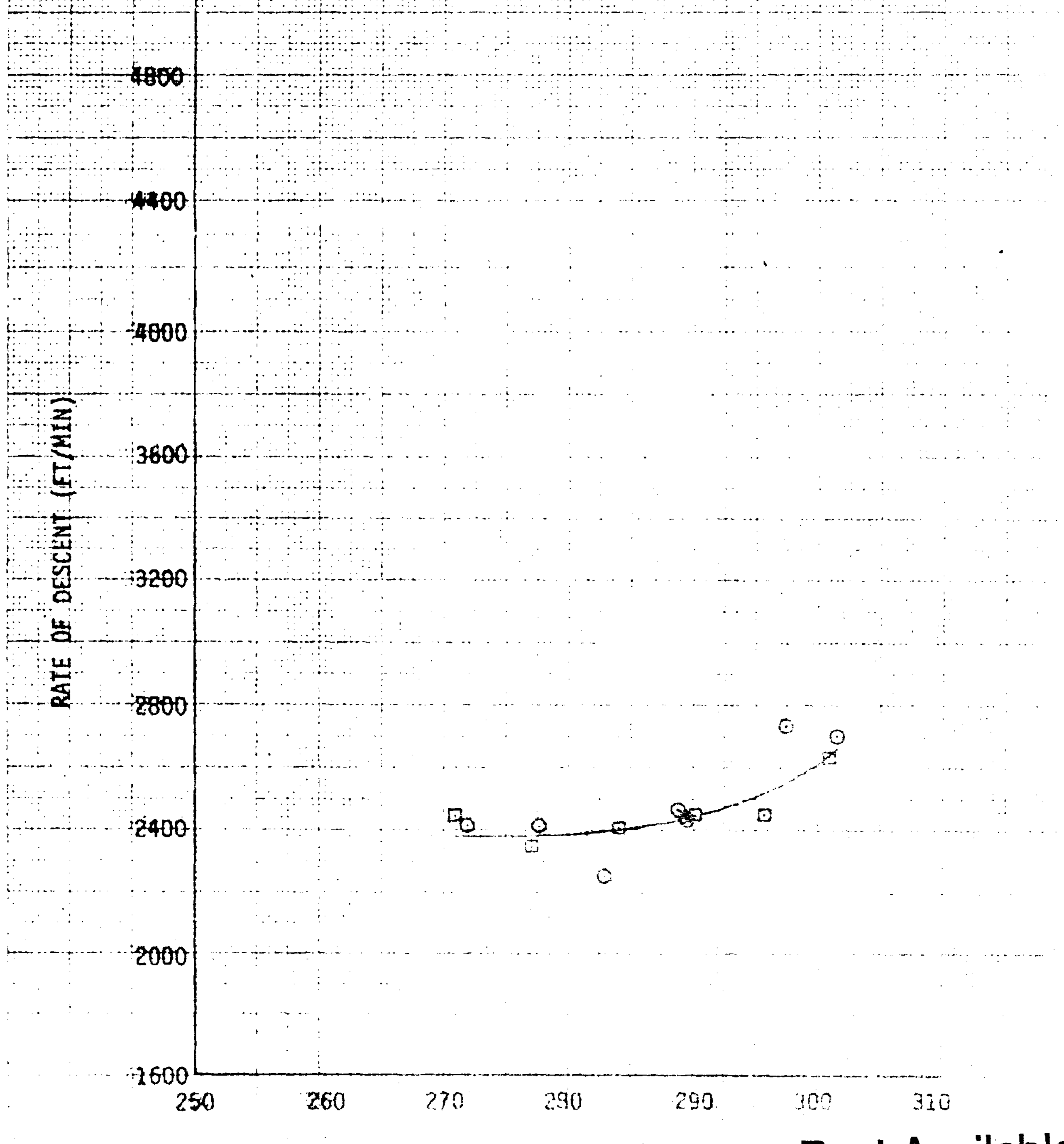

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FIGURE 46
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
14,940	202.0(FWD)	0.8(LT)	4640	21.5	290	LEVEL	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT
3. SHADED SYMBOLS DENOTE DIVES AT IRP

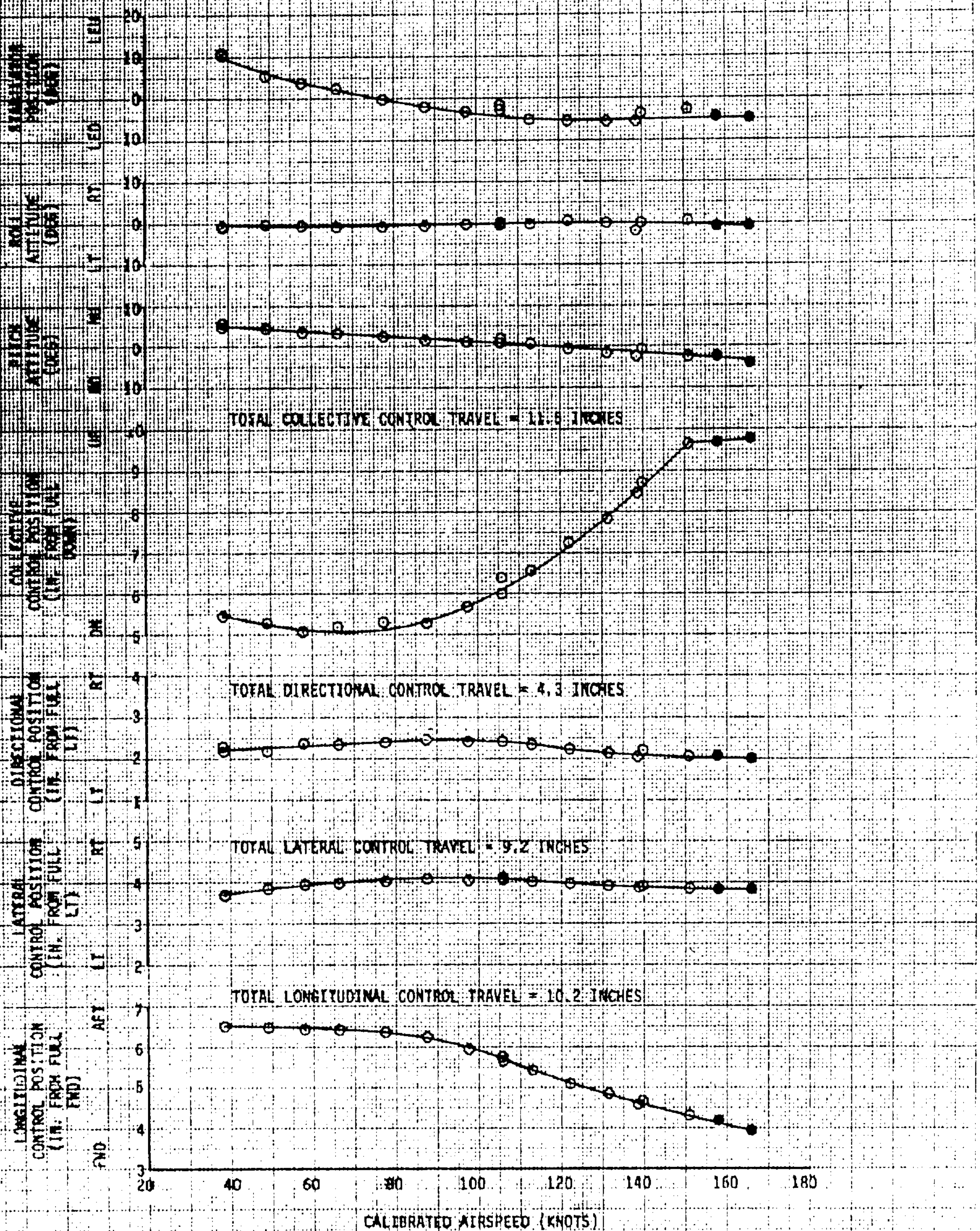

FIGURE 46
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
15,520	206.0(AFT)	0.8(LT)	14,920	4.5	290	LEVEL	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT
3. SHADED SYMBOLS DENOTE DIVES AT IRP

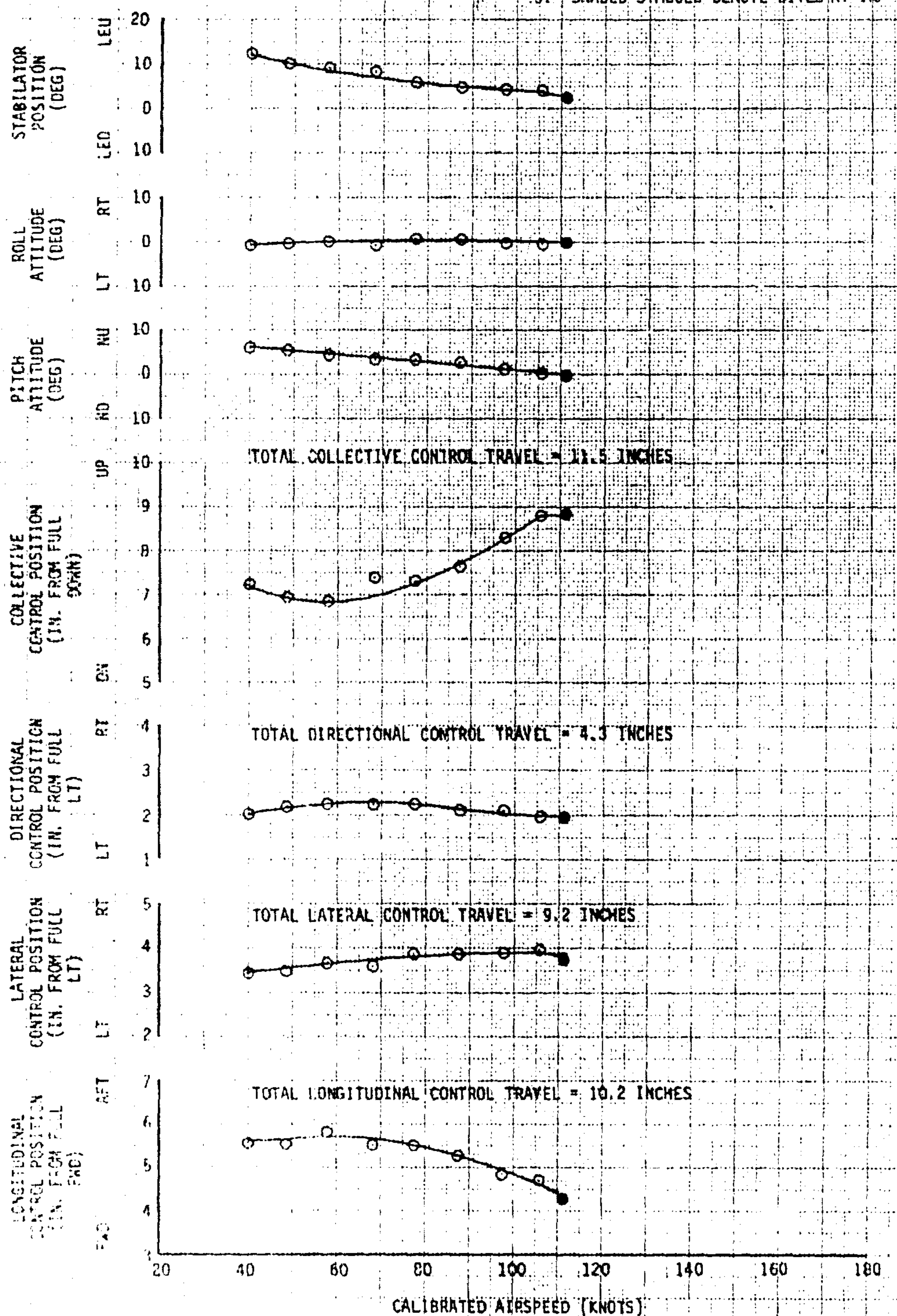

FIGURE 47
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	BASE CONDITION
○	15,200	205.3(MID)	0.8(LT)	4960	20.0	289	AUTOROTATION	ON
□	15,440	204.5(MID)	0.8(LT)	10,180	12.0	289	AUTOROTATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT

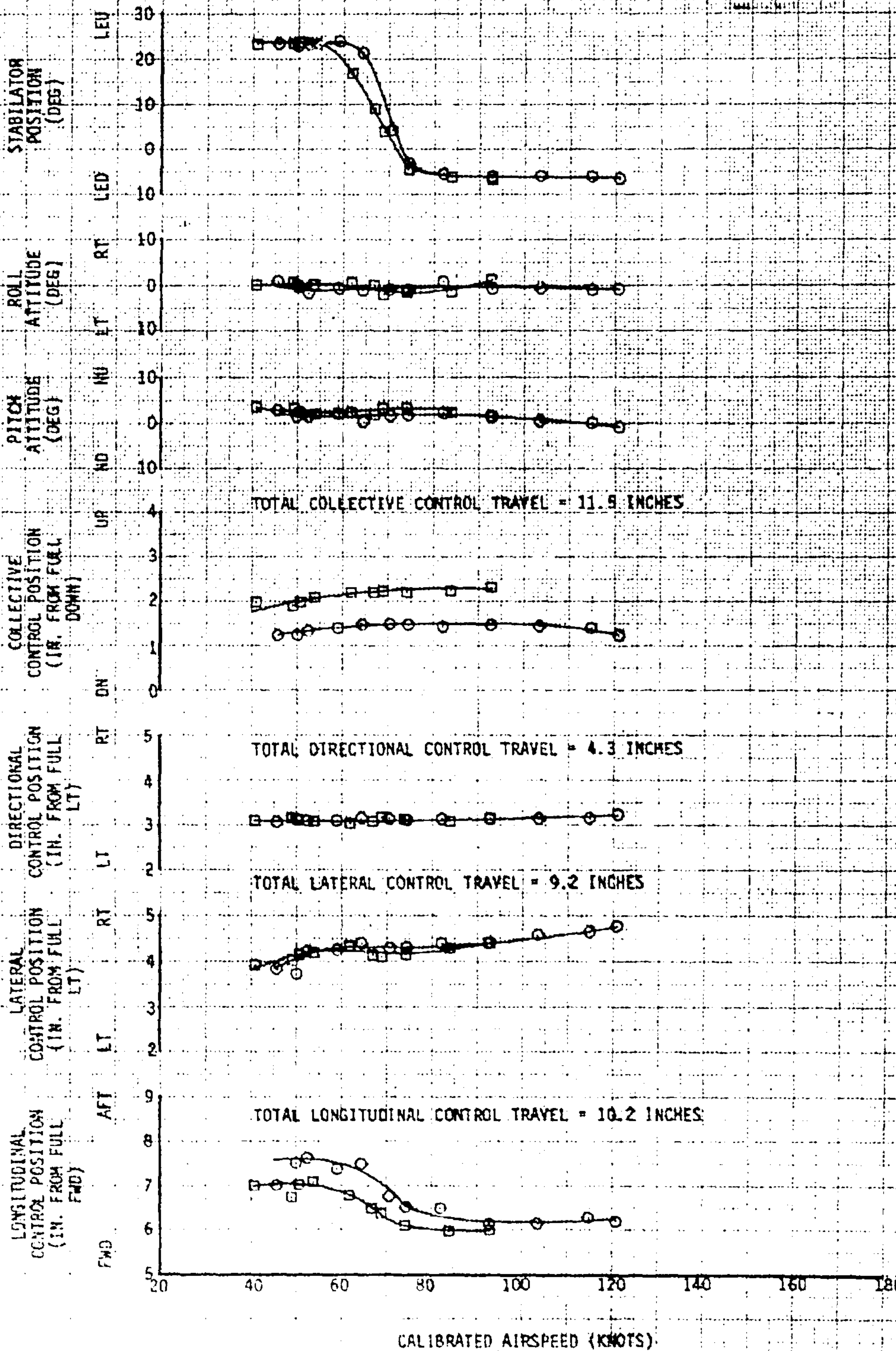

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FIGURE 48
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
○	16,560	204.5(MID)	0.8(LT)	5080	25.0	290	AUTOROTATION	ON
□	16,700	204.4(MID)	0.8(LT)	9900	13.0	293	AUTOROTATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT

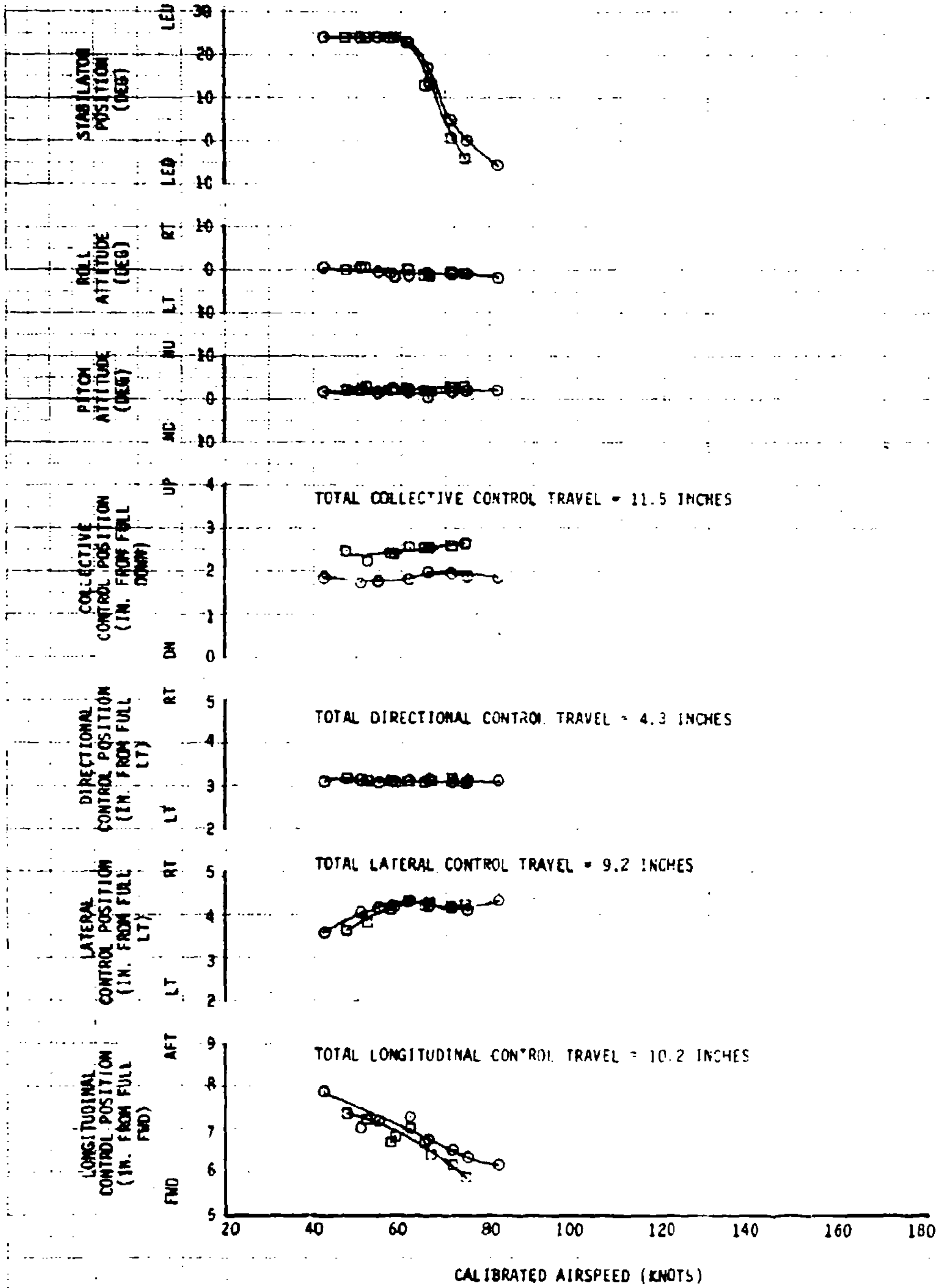


FIGURE 49
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

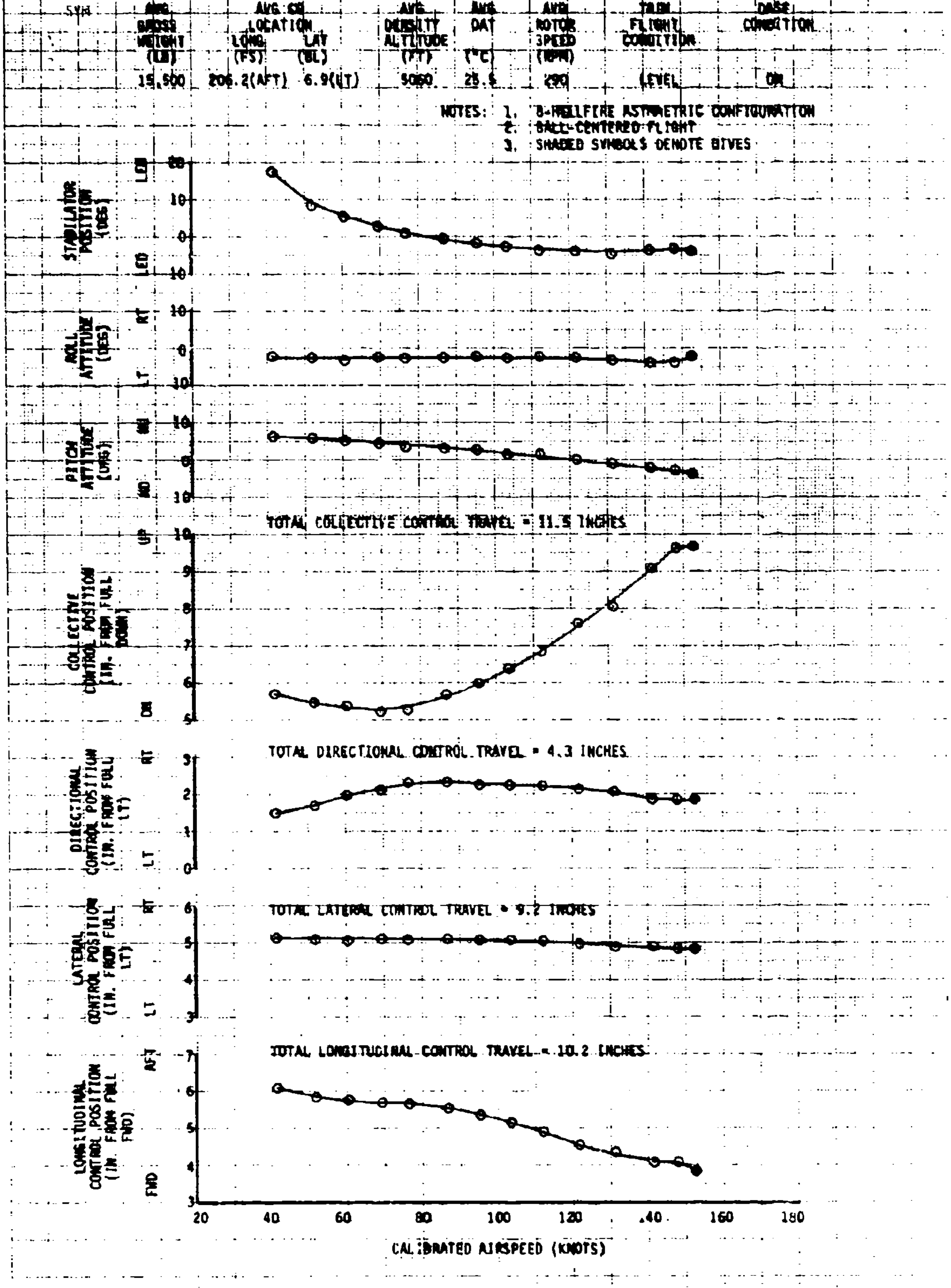
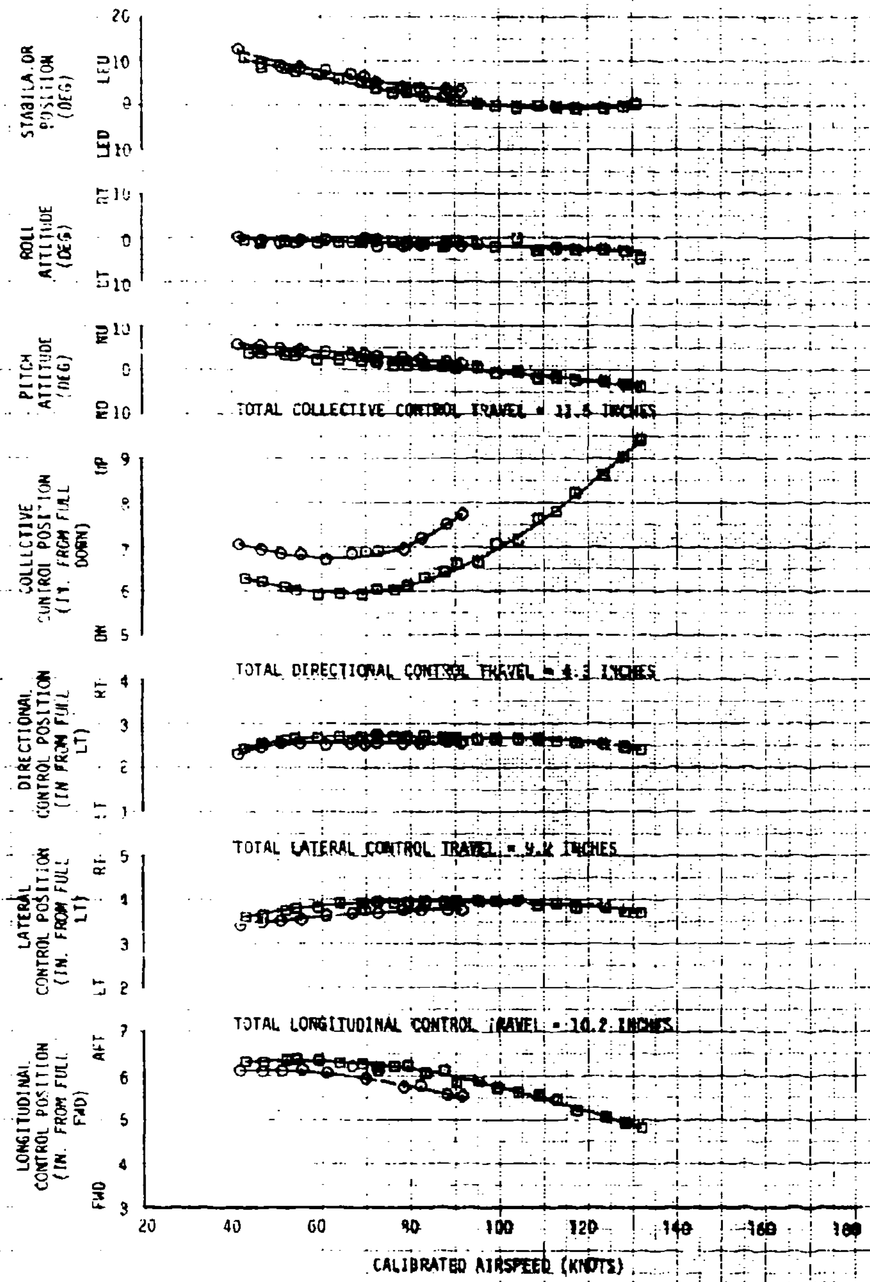


FIGURE 50
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
○	14,680	202.2(FWD)	0.8(LT)	15,560	0.5	291	LEVEL	ON
□	14,840	202.2(FWD)	0.8(LT)	11,120	13.5	290	LEVEL	ON

NOTES: 1. 2-EXTERNAL FERRY TANK CONFIGURATION
2. ZERO SIDESLIP FLIGHT



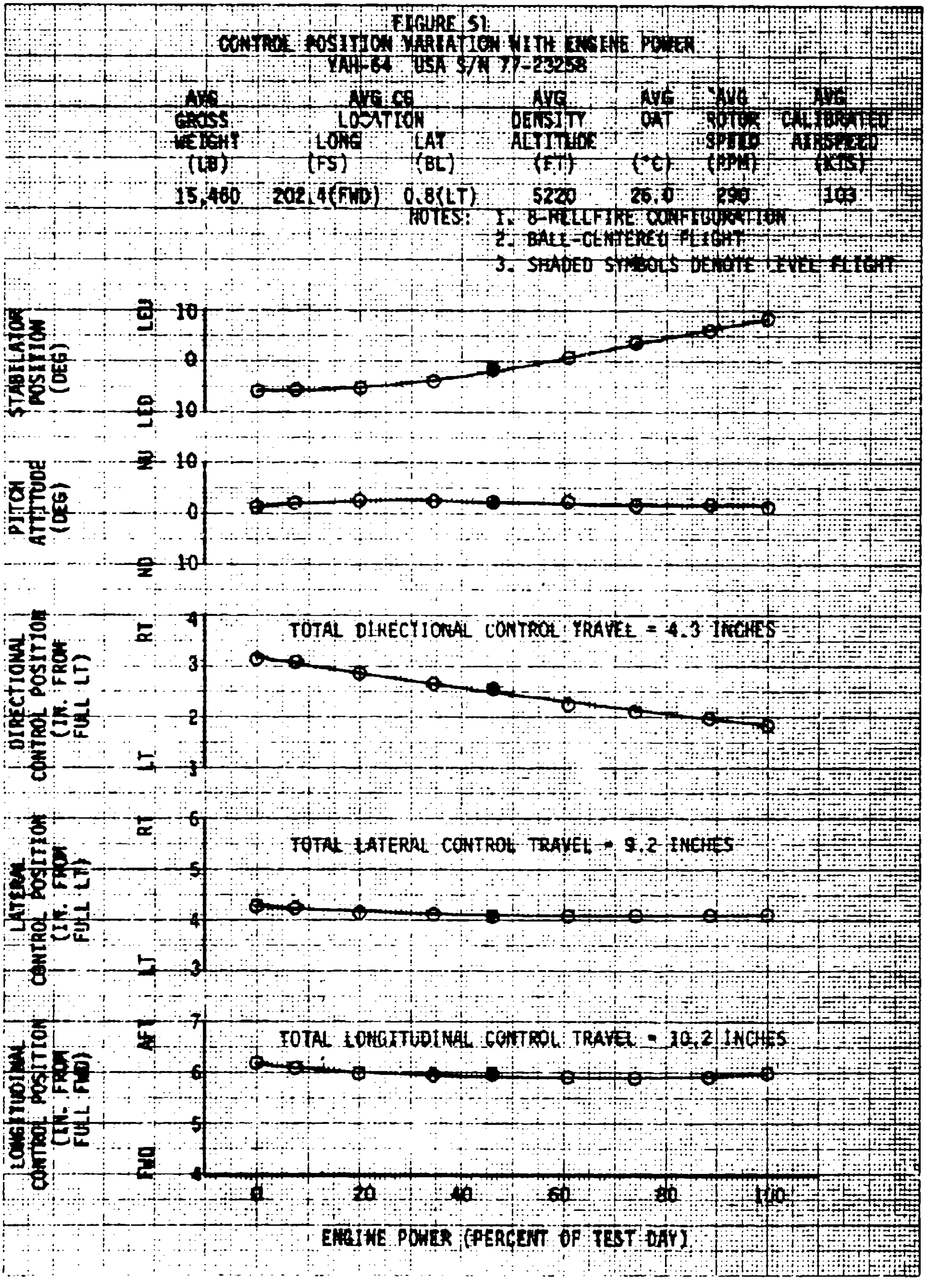


FIGURE 51
CONTROL POSITION VARIATION WITH ENGINE POWER
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 15,460
AVG CG LOCATION LONG (FS) 202.4(FWD) LAT (BL) 0.8(LT)
AVG DENSITY ALTITUDE (FT) 5220
AVG OAT (°C) 26.0
AVG ROTOR SPEED (RPM) 290
AVG CALIBRATED AIRSPEED (KTS) 103
NOTES: 1. 8-HELLFIRE CONFIGURATION
2. BALL-CENTERED FLIGHT
3. SHADED SYMBOLS DENOTE LEVEL FLIGHT
STABILATOR POSITION (DEG)
TED
LED
PITCH ATTITUDE (DEG)
NU
ND
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL DIRECTIONAL CONTROL TRAVEL = 4.3 INCHES
RT
LT
LATERAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL LATERAL CONTROL TRAVEL = 9.2 INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.2 INCHES
AFT
FWD
ENGINE POWER (PERCENT OF TEST DAY)

FIGURE 52
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
●	15,300	202.2(FWD)	-0.8	4720	27.0	290	LEVEL	ON
■	14,920	202.2(FWD)	-0.8	4820	26.5	290	LEVEL	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

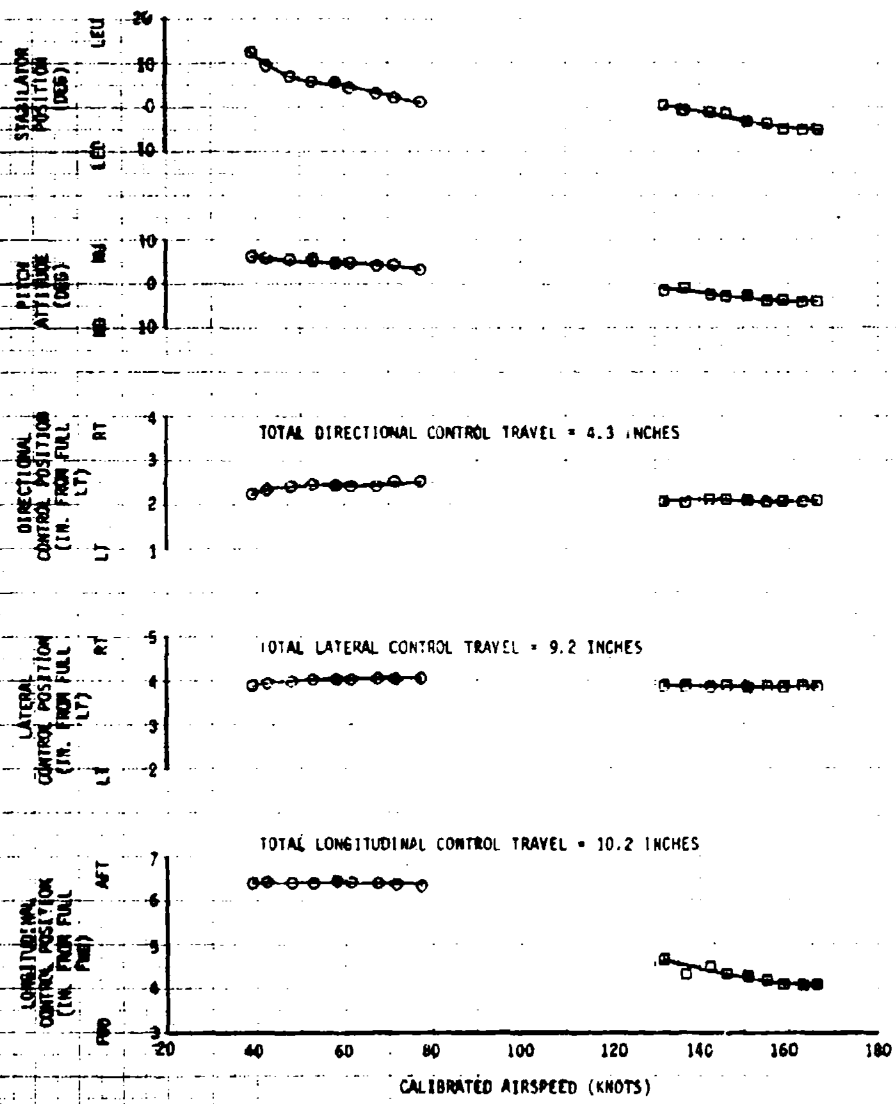


FIGURE 53
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	BASE CONDITION
○	15,300	206.0(AFT)	0.8(LT)	15,300	5.0	290	LEVEL	ON
□	15,160	206.0(AFT)	0.8(LT)	14,780	4.0	290	LEVEL	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

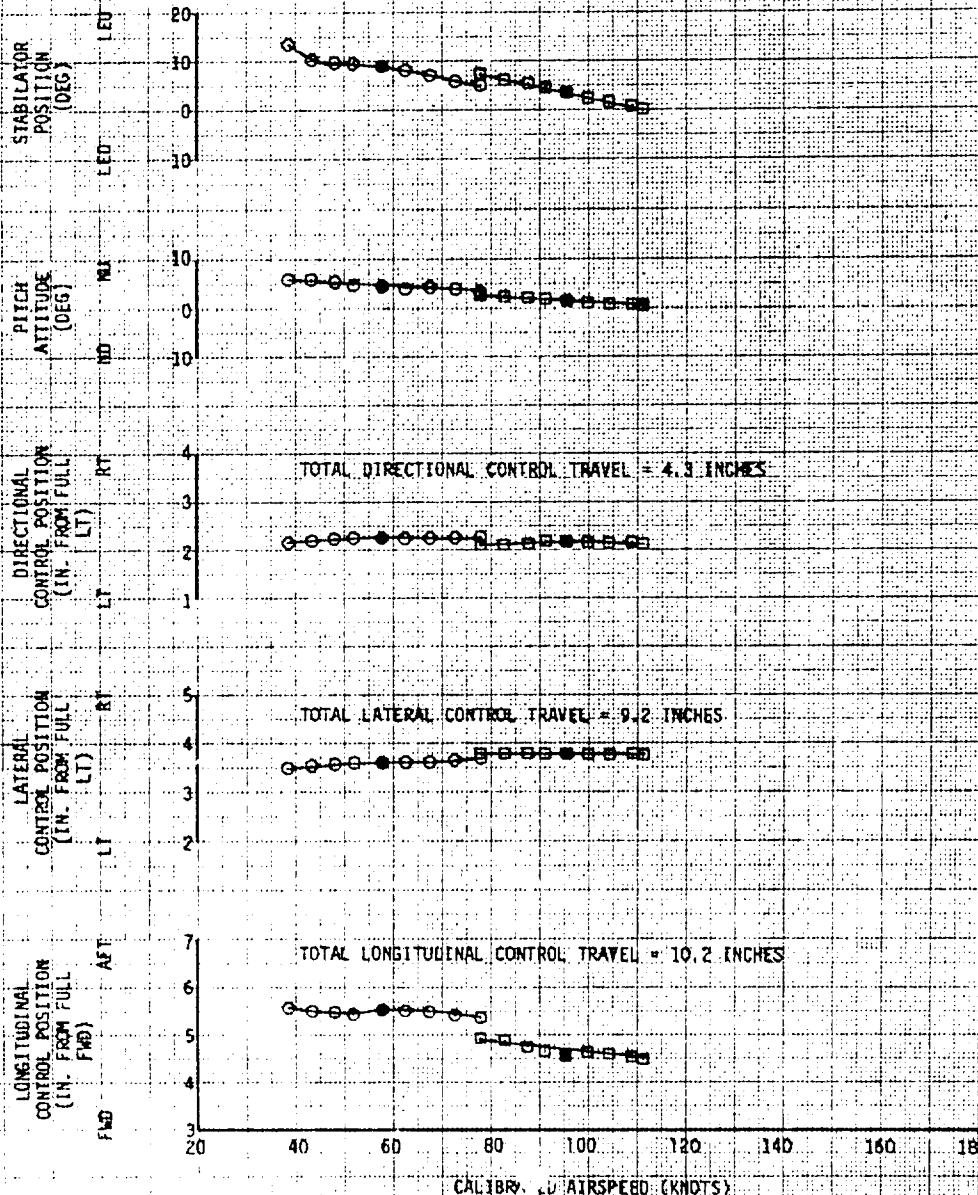


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FIGURE 54
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	BASE CONDITION
○	15,760	206.2(AFT)	6.8(LT)	4820	26.5	290	LEVEL	ON
□	15,120	206.2(AFT)	7.1(LT)	5160	25.0	290	LEVEL	ON

NOTES: 1. 8-HELLFIRE ASYMMETRIC CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

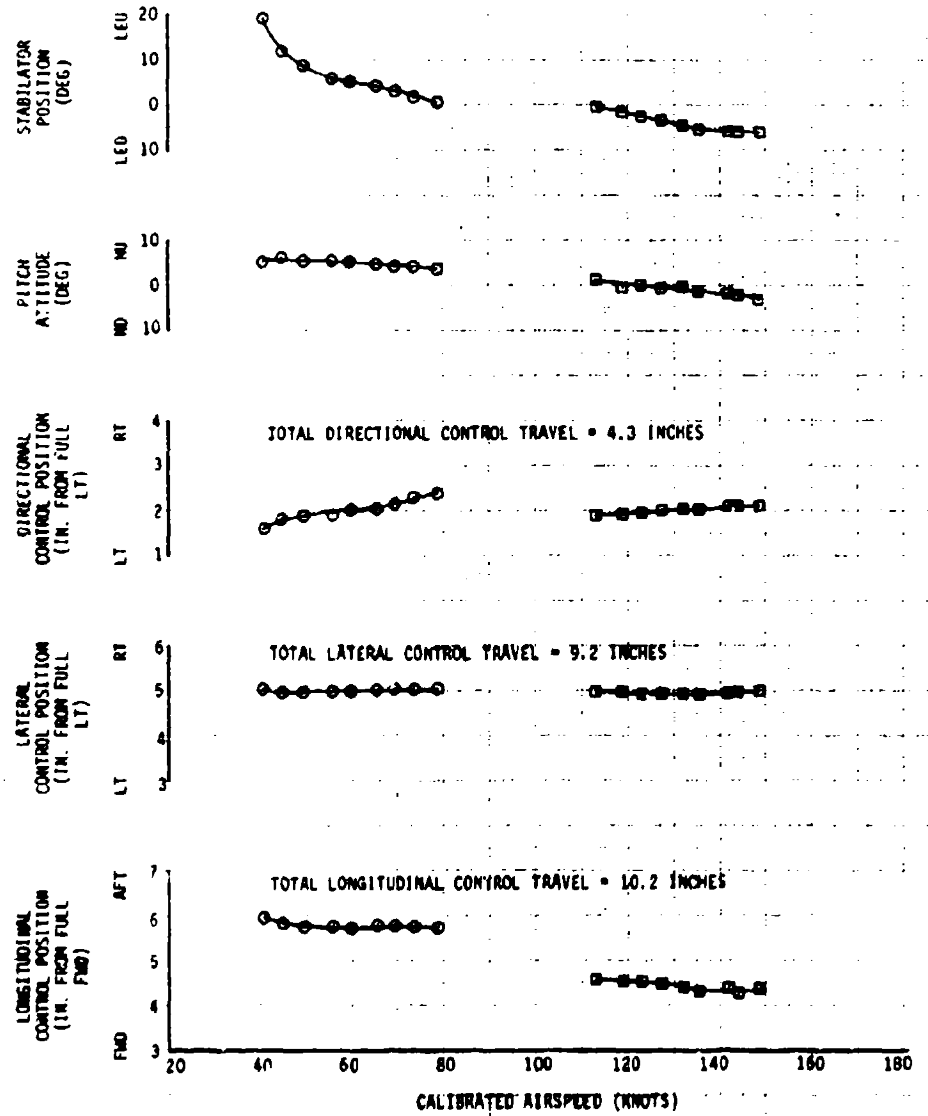

FIGURE 55
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)	TRIM FLIGHT CONDITION	BASE CONDITION
⊙	15,840	202.5(FWD)	0.8(LT)	5040	26.5	290	60	LEVEL	ON
⊡	15,000	202.5(FWD)	0.8(LT)	5220	25.0	290	136	LEVEL	ON

NOTES: 1. B-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

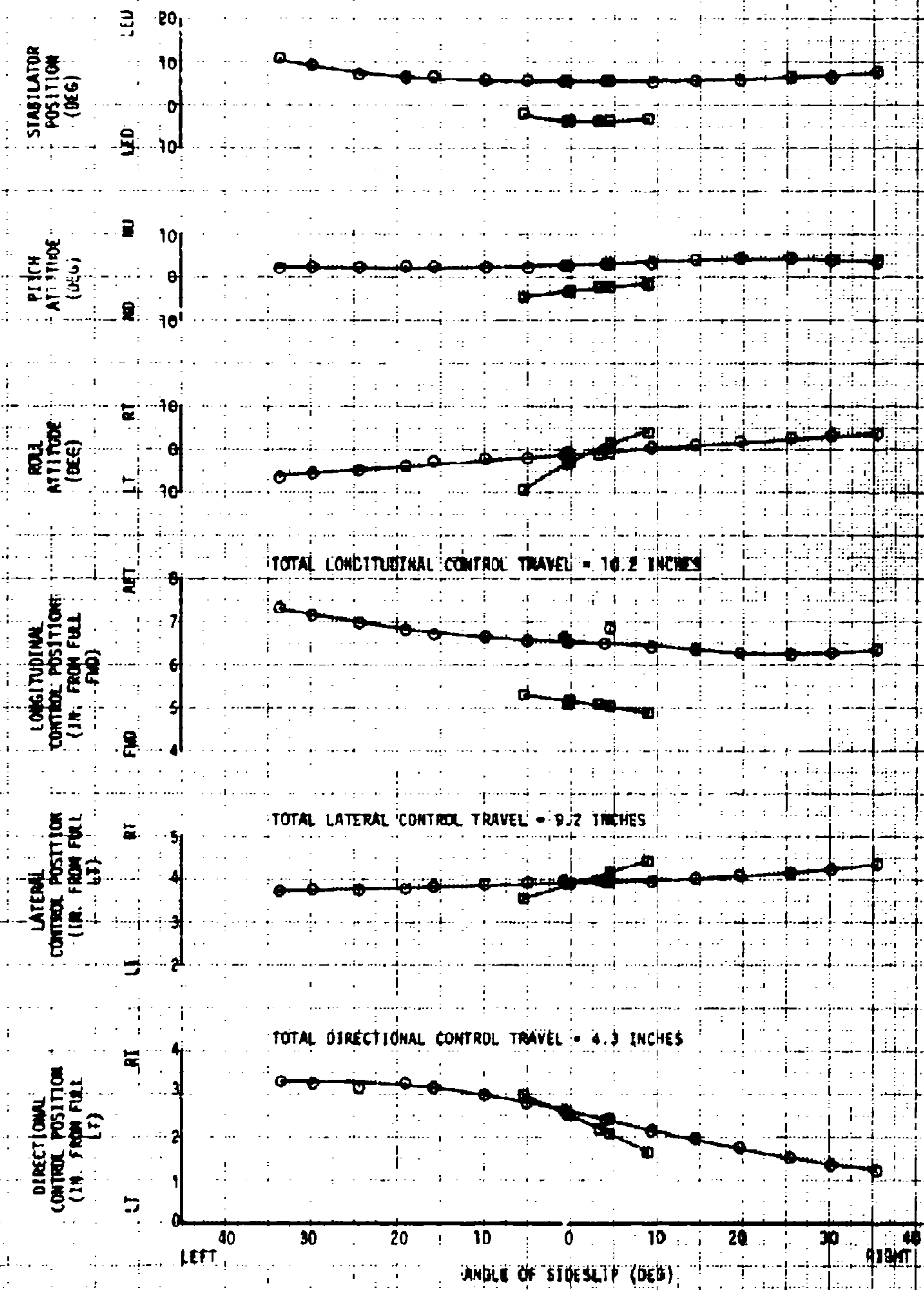

FIGURE 56
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)	TRIM FLIGHT CONDITION	BASE CONDITION
○	14,940	206.0(AFT)	0.8(LT)	15,000	3.0	290	60	LEVEL	ON
□	14,520	206.0(AFT)	0.8(LT)	14,500	3.5	290	95	LEVEL	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

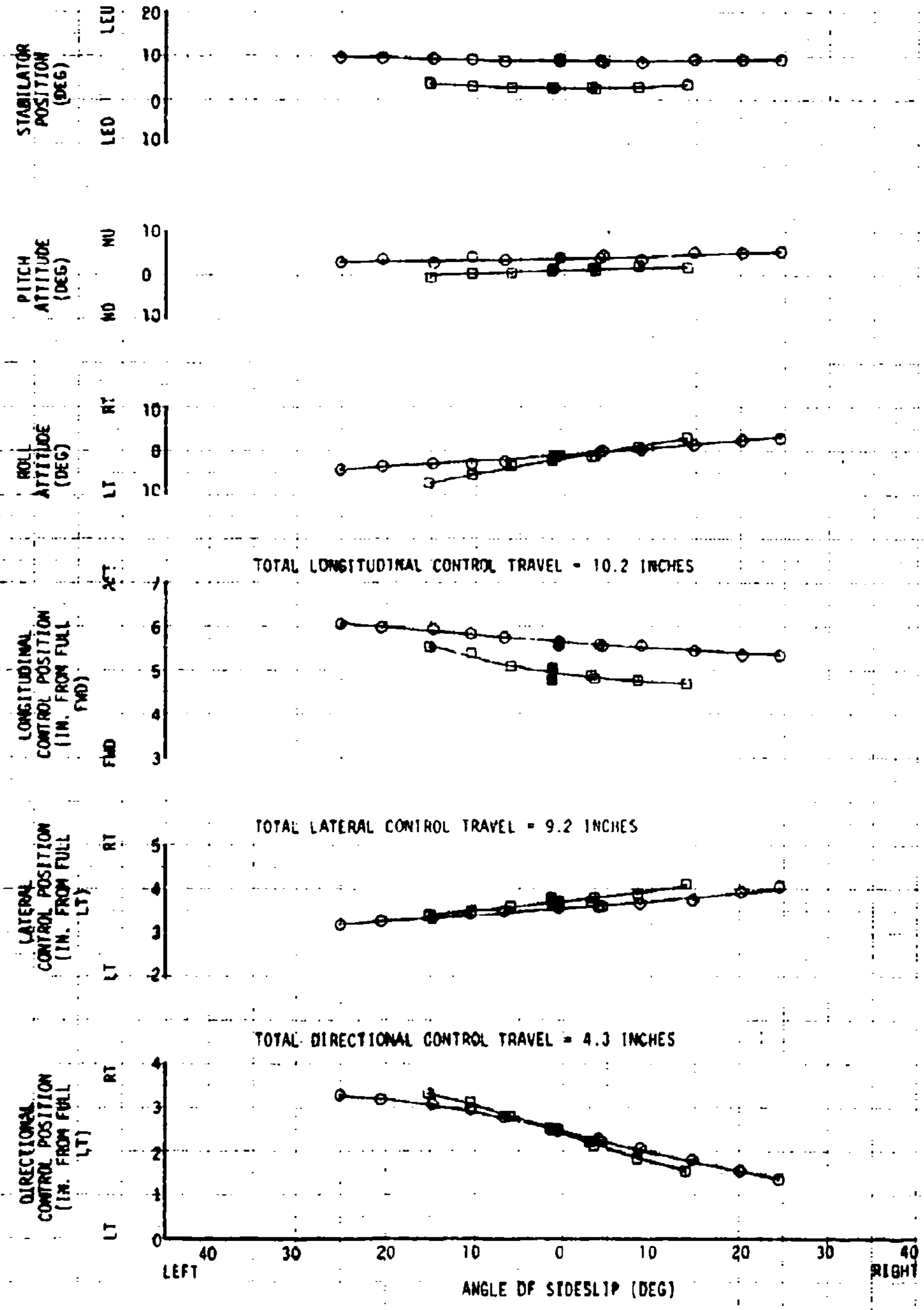

FIGURE 57
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	DASE CONDITION
○	17,720	204.0(AFT)	0.8(LT)	5360	26.0	290	60	LEVEL	ON
□	17,640	204.1(AFT)	0.8(LT)	4760	24.5	290	137	LEVEL	ON

NOTES: 1. 16-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

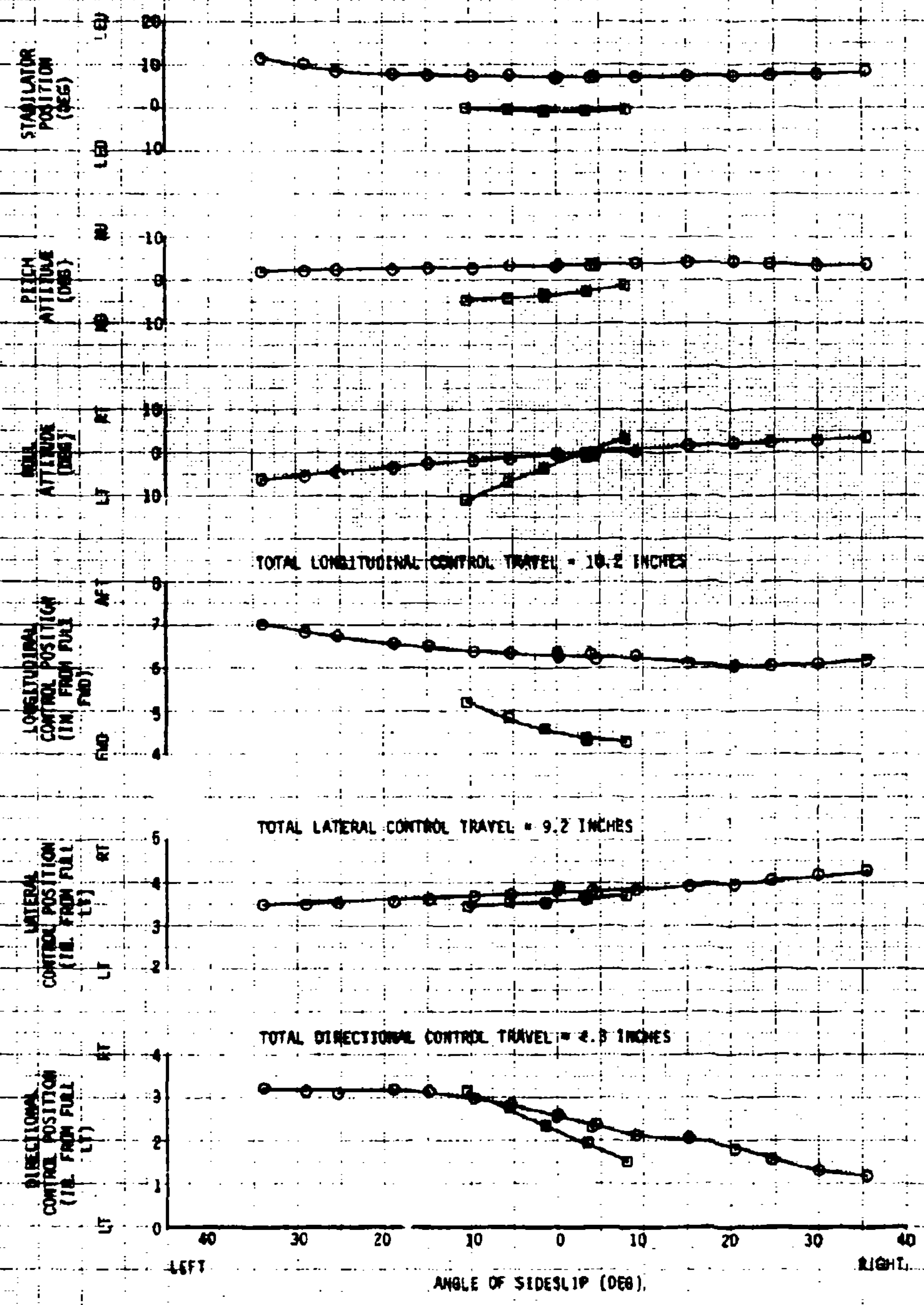

FIGURE 57
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)	TRIM FLIGHT CONDITION	DASE CONDITION
○	17,720	204.0(AFT)	0.8(LT)	5360	26.0	290	60	LEVEL	ON
□	17,640	204.1(AFT)	0.8(LT)	4760	24.5	290	137	LEVEL	ON

NOTES: 1. 16-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

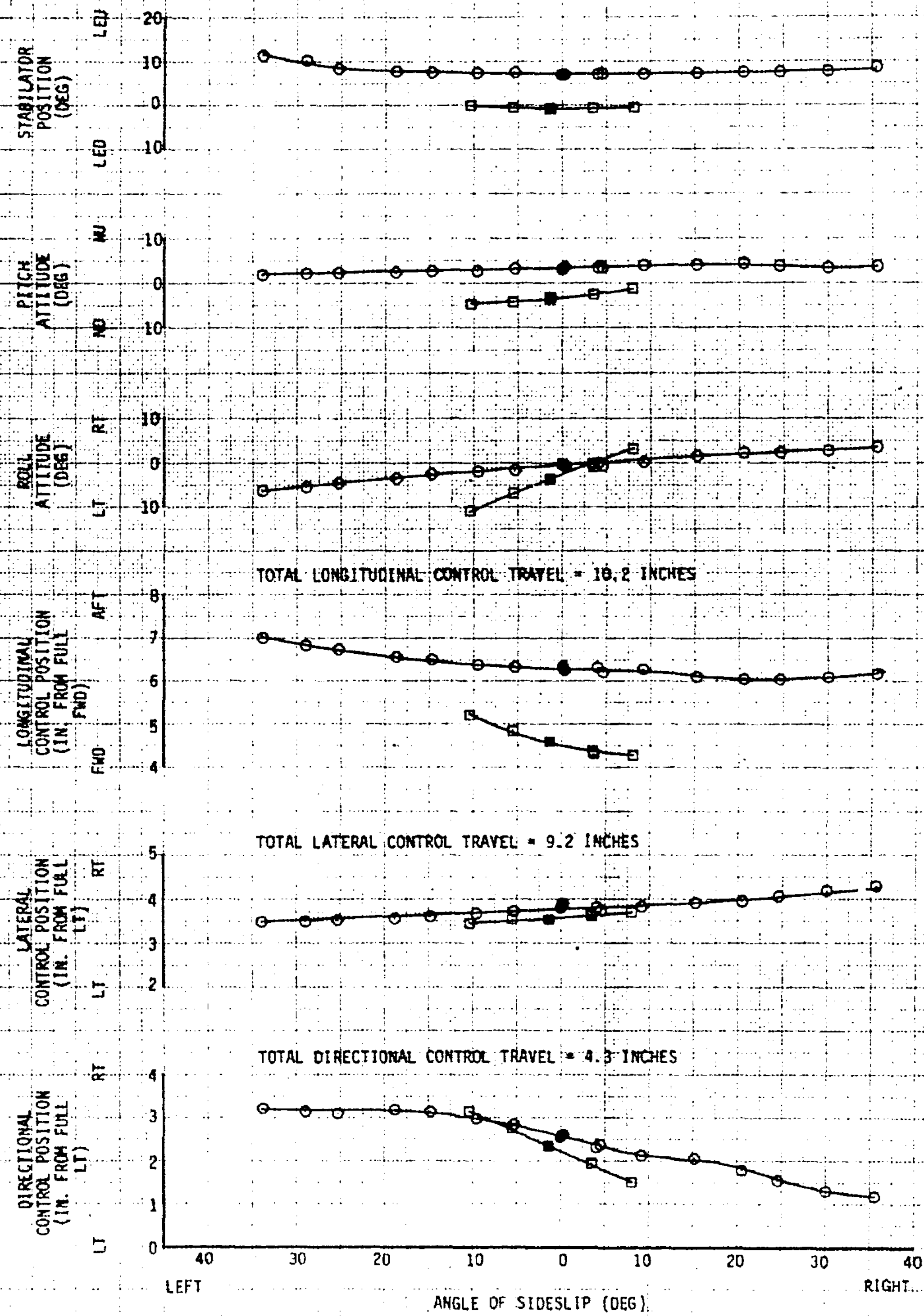

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FIGURE 58
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)	TRIM FLIGHT CONDITION	DASE CONDITION
⊙	15,200	205.6(AFT)	0.9(LT)	5100	29.0	290	60	LEVEL	ON
⊡	14,860	206.3(AFT)	0.9(LT)	5120	25.0	290	134	LEVEL	ON

NOTES: 1. 8-HELLFIRE ASYMMETRIC CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

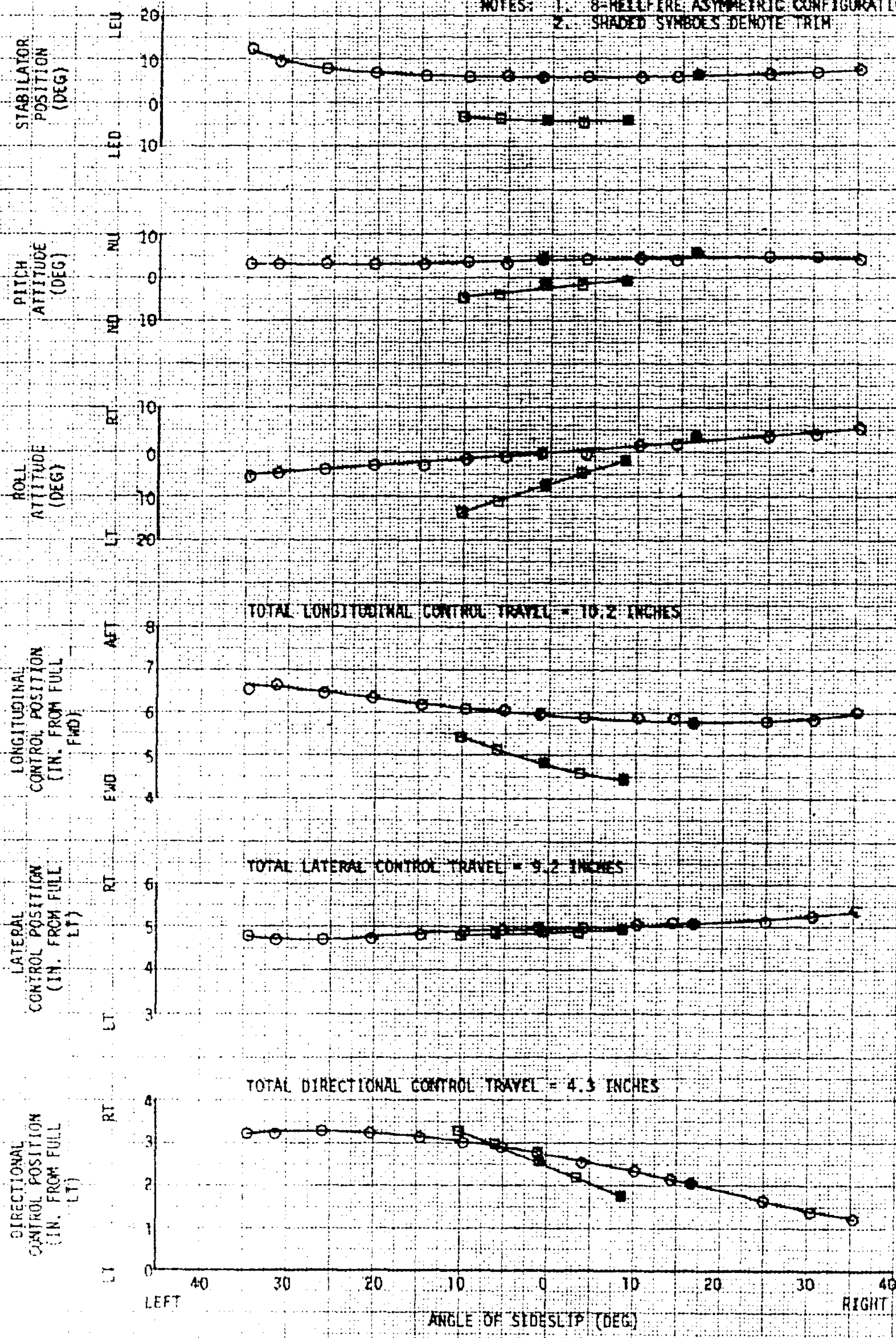

FIGURE 59
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
⊙	15,780	202.0(FWD)	0.8(LT)	5220	26.0	290	LT TURN
⊡	15,540	202.0(FWD)	0.8(LT)	6520	25.0	290	RT TURN

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. TRIM AIRSPEED = 139 KCAS
3. DASE ON, ATTITUDE HOLD OFF
4. SHADED SYMBOL DENOTES TRIM
5. BALL-CENTERED FLIGHT

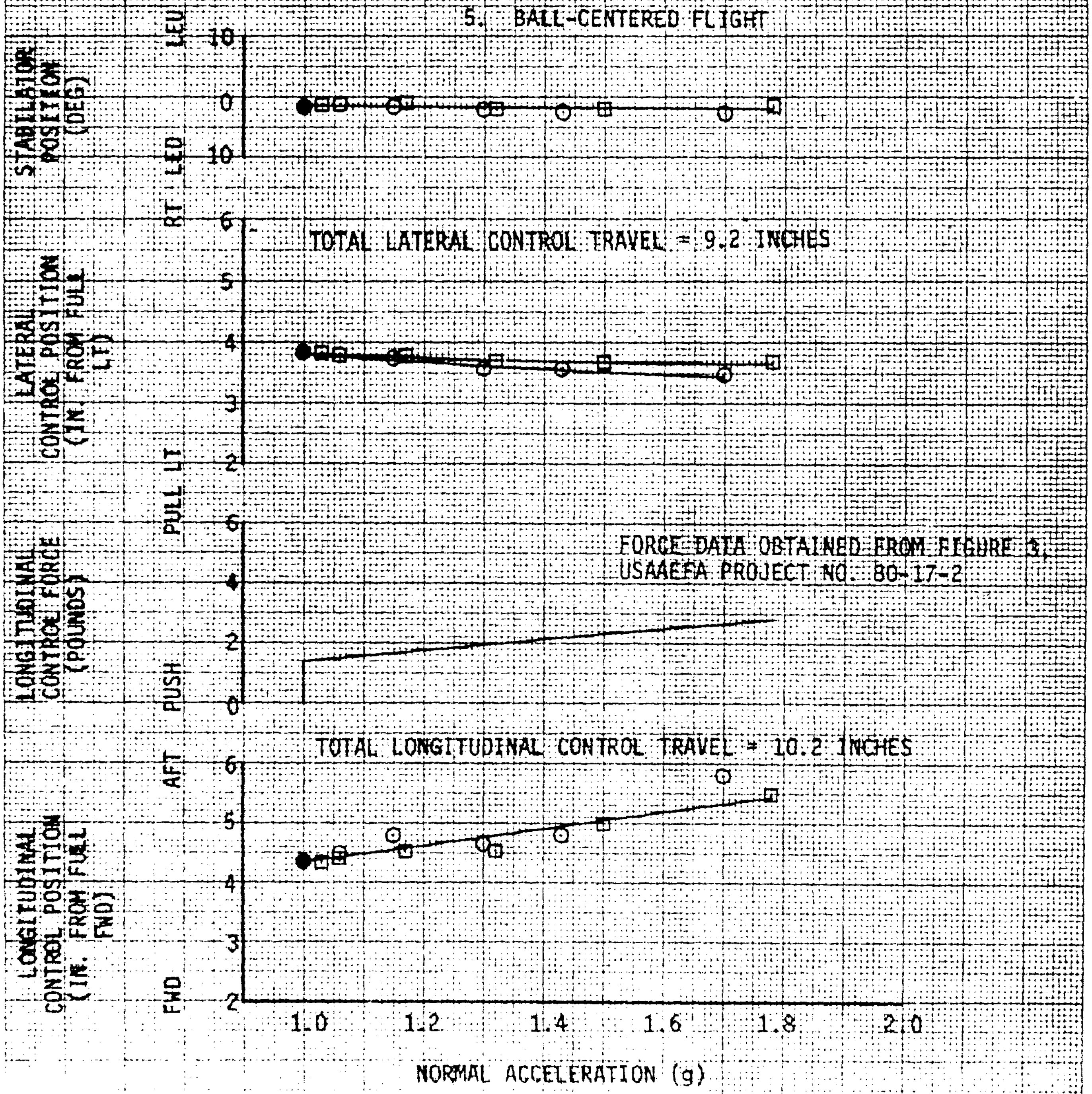

FIGURE 60
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	15,680	206.2(AFT)	7.2(LT)	5120	25.0	290	LT TURN
□	15,420	205.5(AFT)	6.9(LT)	7260	25.5	290	RT TURN

NOTES:
1. 8-HELLFIRE ASYMMETRIC CONFIGURATION
2. TRIM AIRSPEED = 136 KCAS
3. DASE ON, ATTITUDE HOLD OFF
4. SHADED SYMBOL DENOTES TRIM
5. BALL-CENTERED FLIGHT

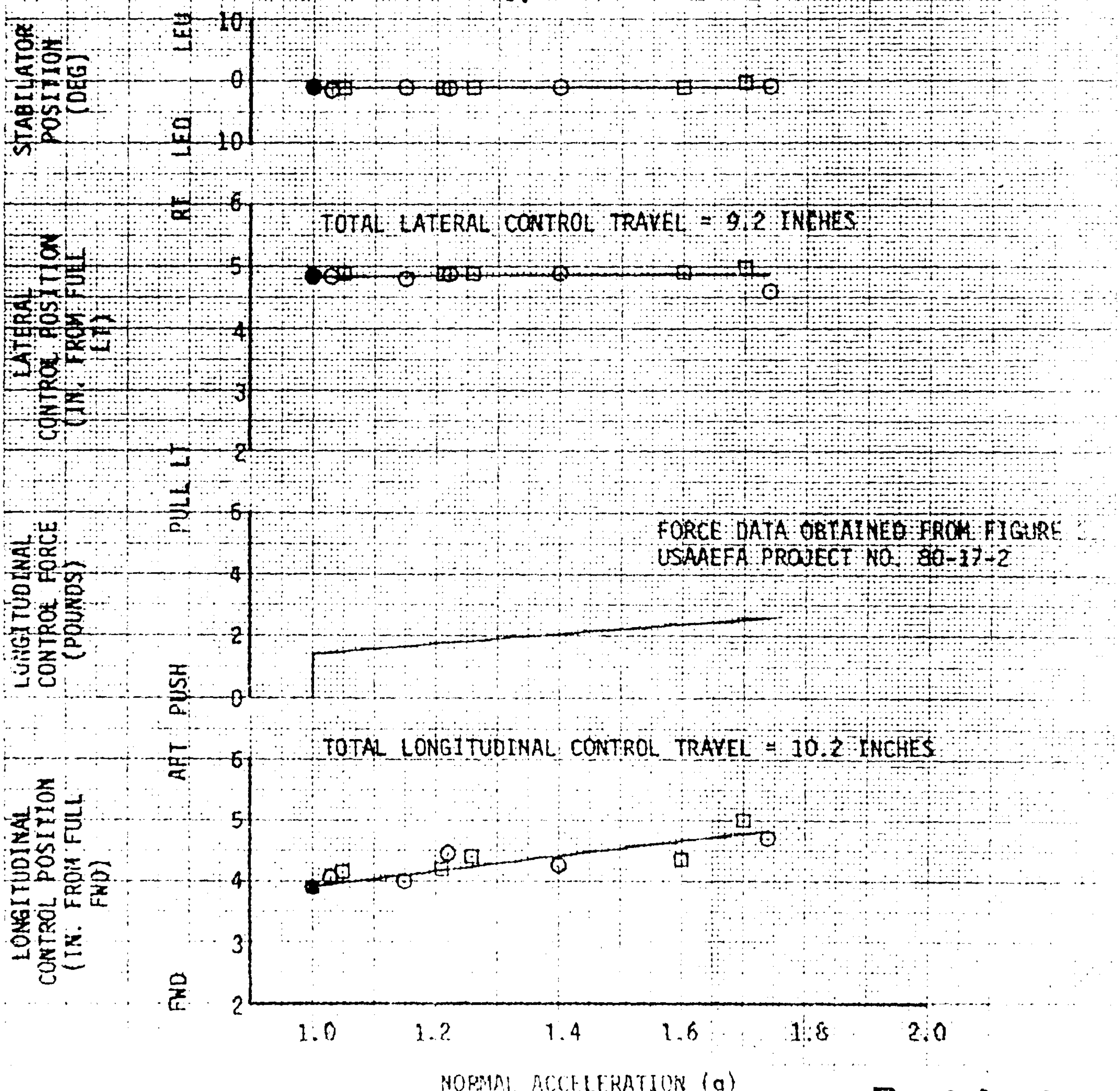

FIGURE 62

LONGITUDINAL LONG-TERM RESPONSE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
17,720	204.2(AFT)	0.8(LT)	5500	25.0	290	LEVEL	ON	91

NOTE: 16-HELLFIRE CONFIGURATION

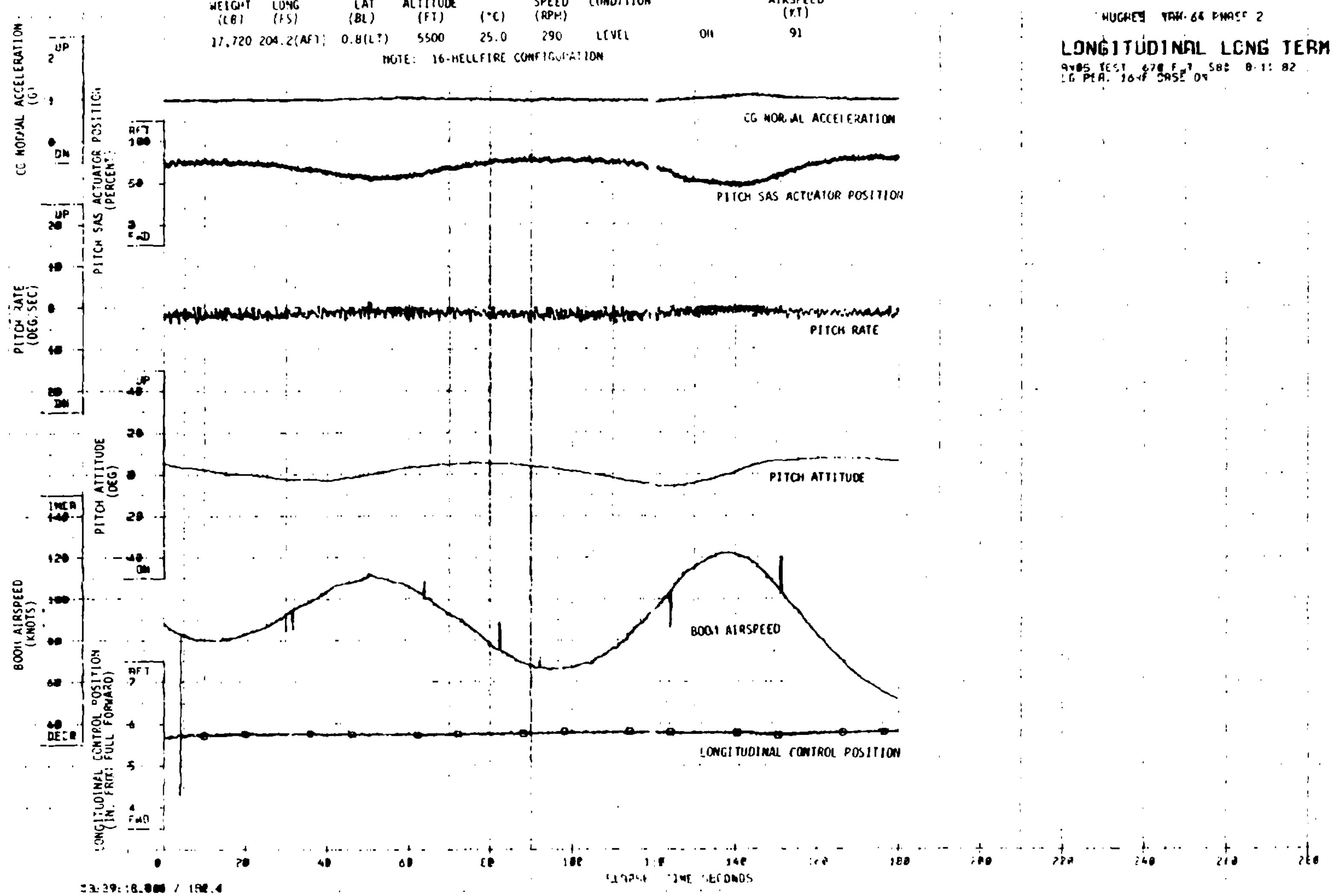

FIGURE 63

LONGITUDINAL LONG TERM RESPONSE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,180	204.5(MID)	0.8(LT)	5220	29.5	291	LEVEL	OFF	96

NOTE: B-HELLFIRE CONFIGURATION

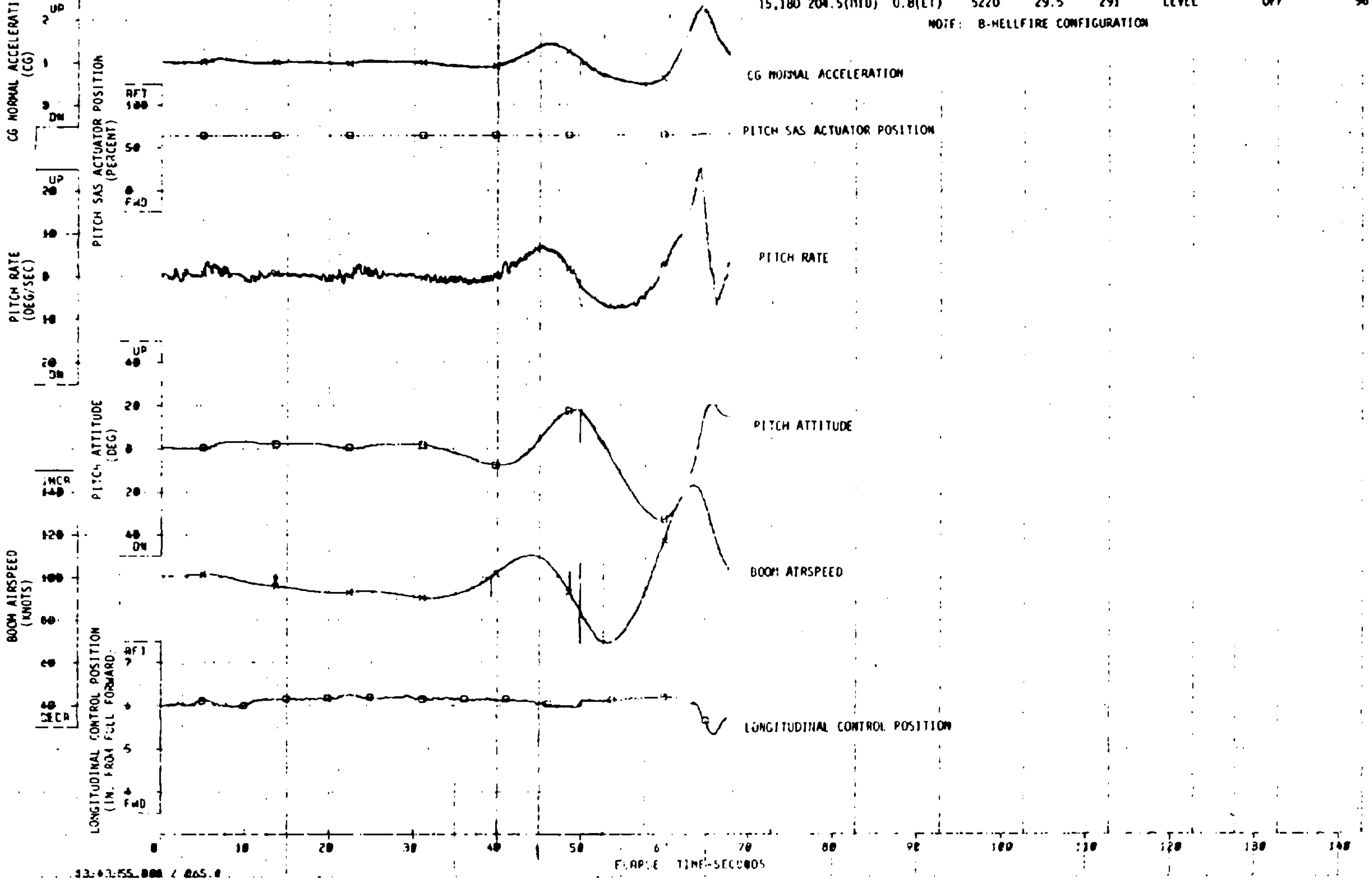

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FIGURE 64

FORWARD CYCLIC PULSE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,360	205.5(AFT)	0.6(LT)	14,680	7.0	290	LEVEL	ON	83

NOTE: 8-HELLFIRE CONFIGURATION

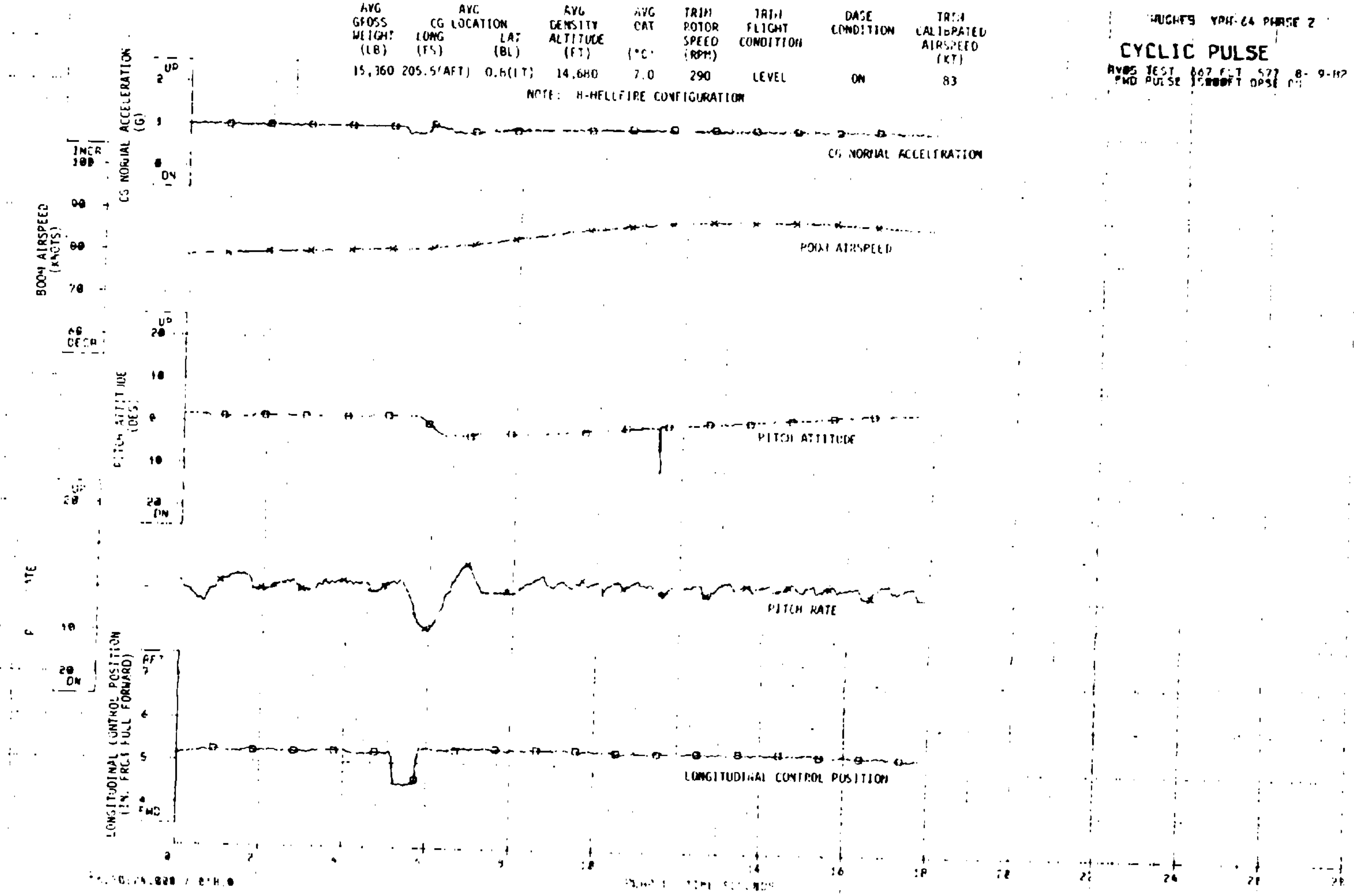

FIGURE 65

AFT CYCLIC PULSE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,360	205.5(AFT)	08.(LT)	14,720	7.0	290	LEVEL	OFF	75

NOTE: 8-HELLFIRE CONFIGURATION

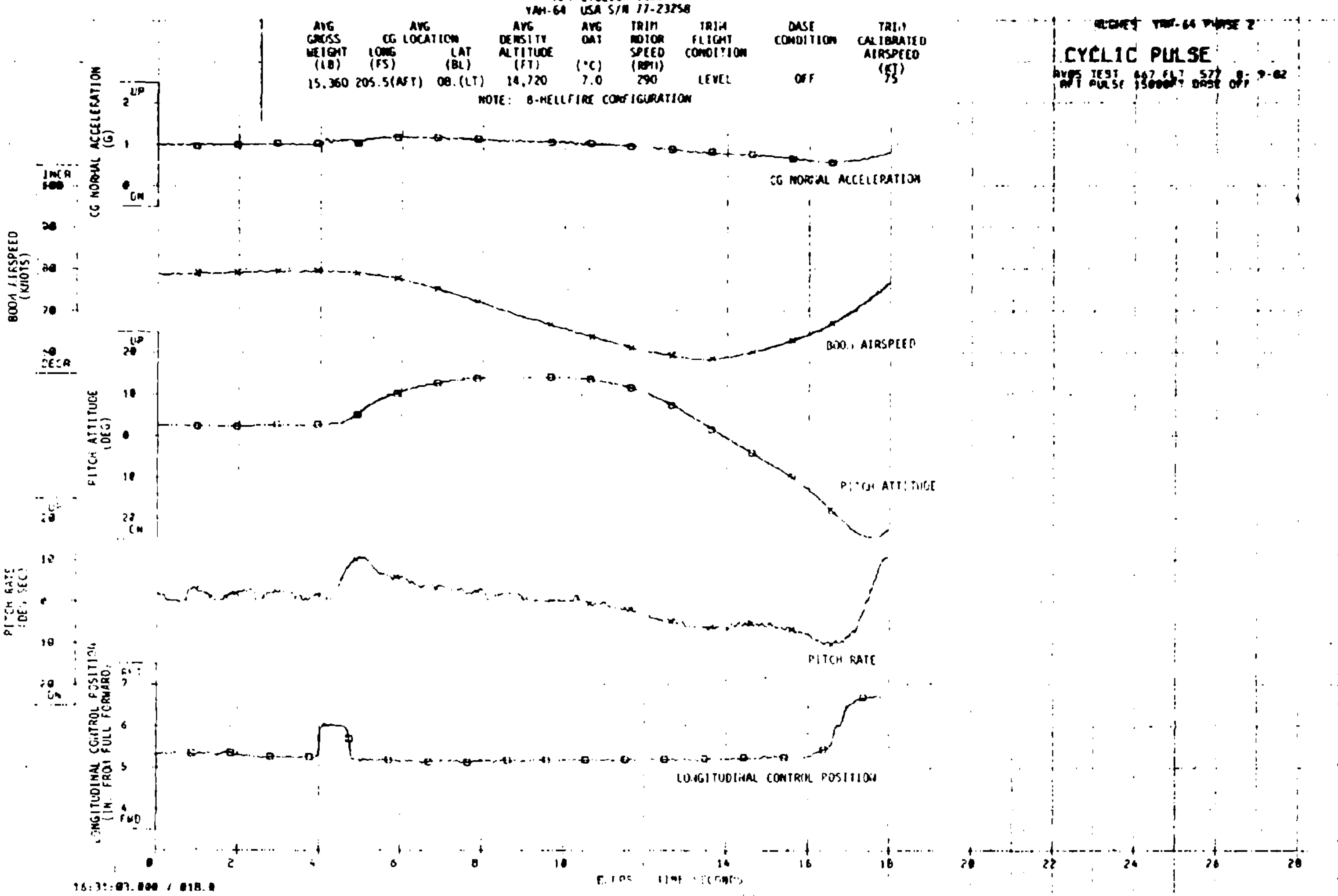

FIGURE 66

PEDAL DOUBLET
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,660	205.5(AFT)	0.8(LT)	14,980	8.0	290	LEVEL	ON	80

NOTE: 8-HELLFIRE CONFIGURATION

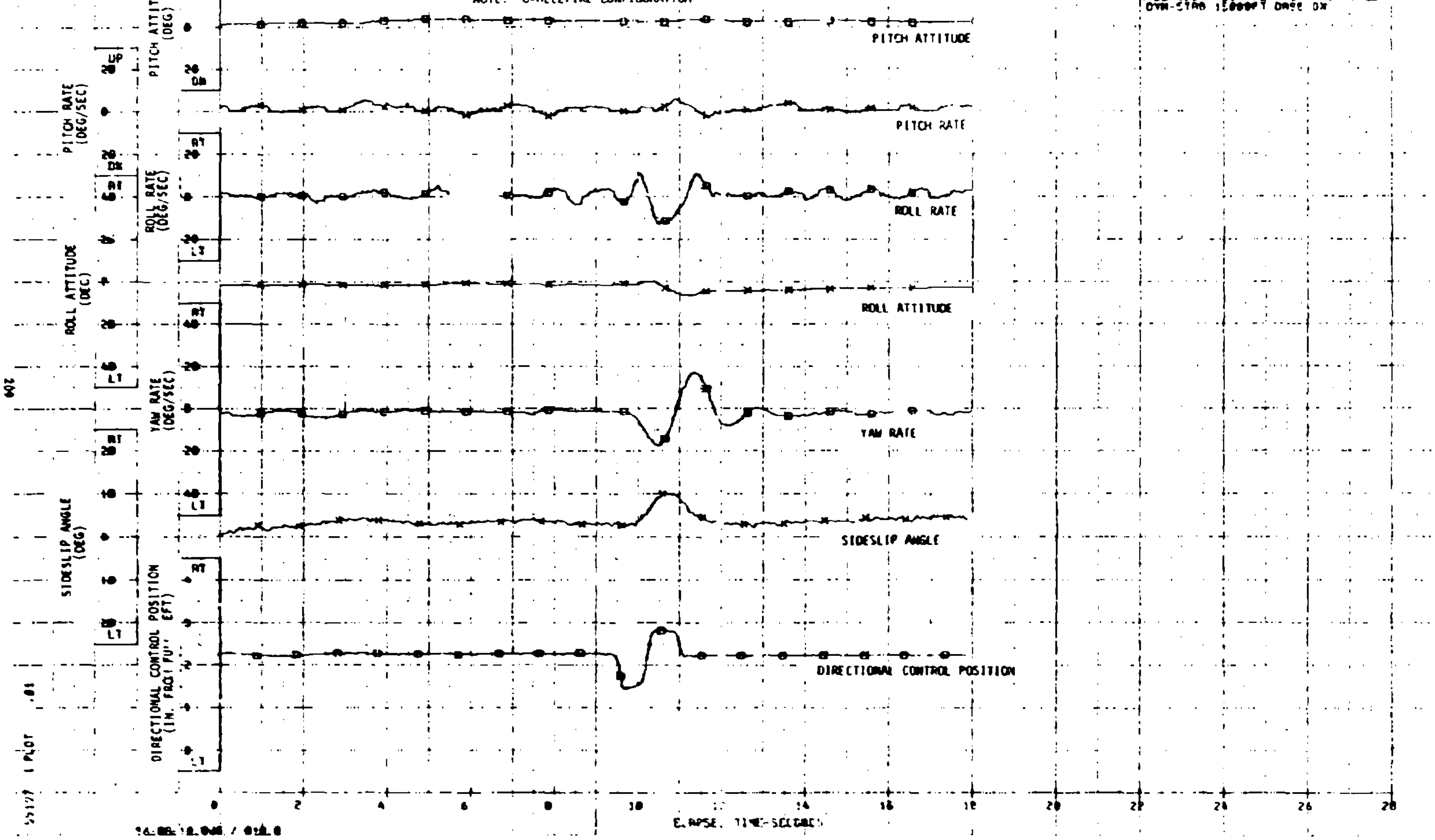

FIGURE 67

PEDAL DOUBLET
YAH-64 USA S/N 77-23256

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,580	205.5(AFT)	0.8(LT)	14,460	6.5	290	LEVEL	OFF	81

NOTE: 8-HELLFIRE CONFIGURATION

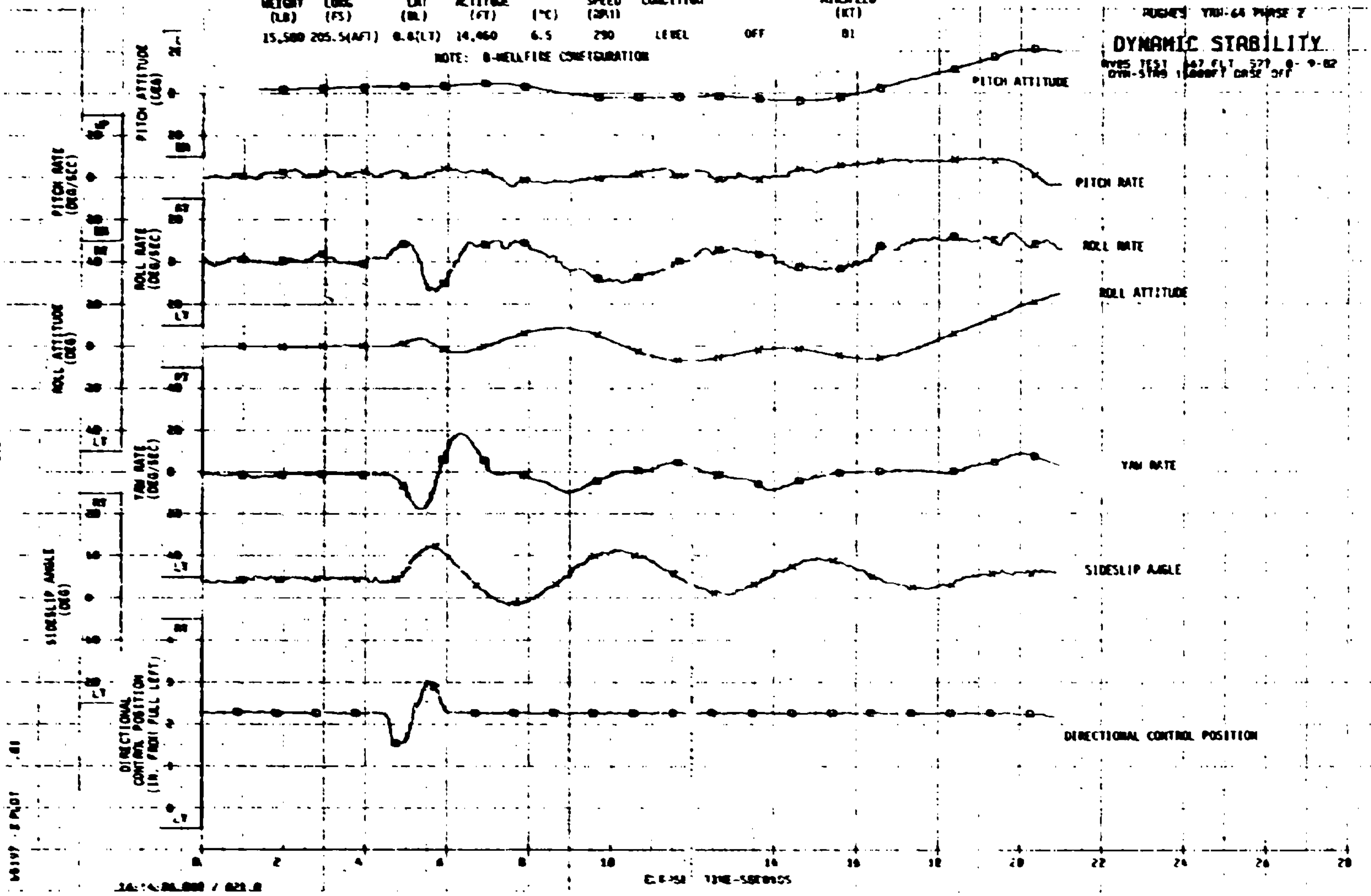

FIGURE 68

PEDAL DOUBLET
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	BASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,864	205.4(RFT)	6.7(LT)	7820	28.0	290	LEVEL	OFF	74

NOTE: 8-HELLFIRE ASYMMETRIC CONFIGURATION

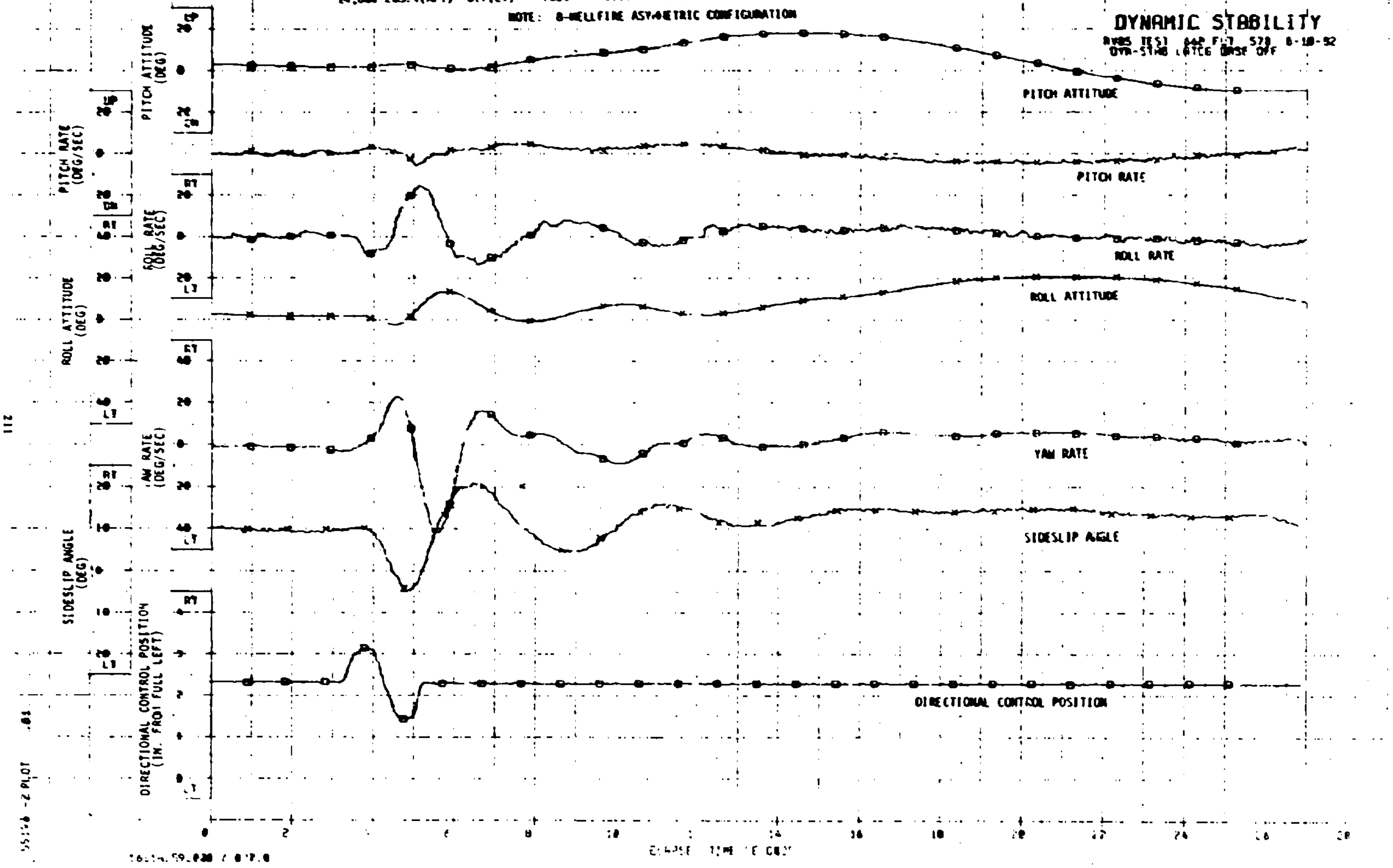

FIGURE 69

PEDAL DOUBLET
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,980	205.4(AFT)	6.7(LT)	5940	28.0	290	LEVEL	ON	79

NOTE: 8-HELLFIRE ASYMMETRIC CONFIGURATION

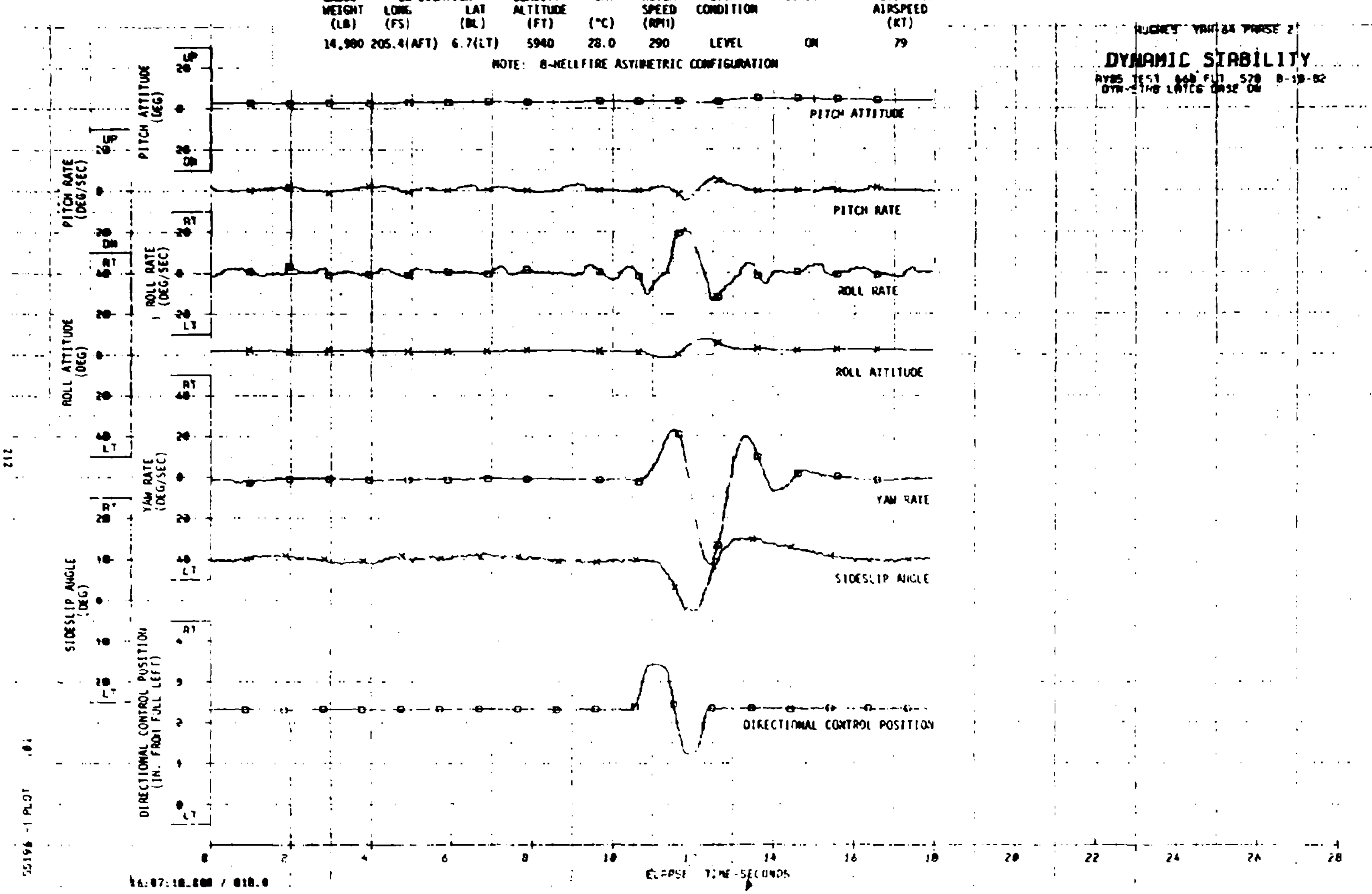

FIGURE 70
LONGITUDINAL CONTROLLABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	DASE CONDITION
○	14,900	205.9(AFT)	0.8(LT)	14,820	5.0	290	79	LEVEL	ON
□	14,780	205.9(AFT)	0.8(LT)	14,660	4.5	290	80	LEVEL	OFF

NOTE: 8-HELLFIRE CONFIGURATION

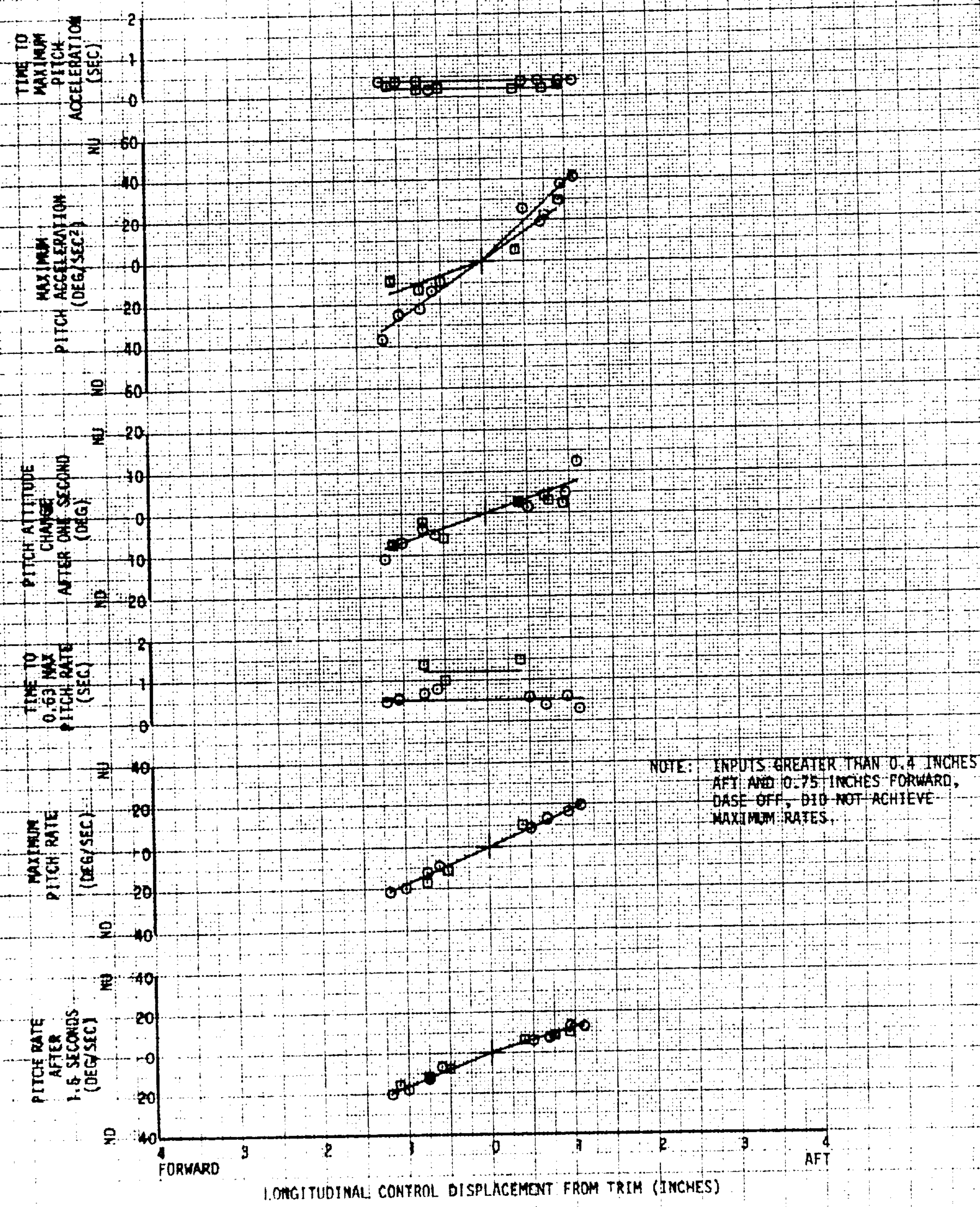

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FIGURE 71

AFT LONGITUDINAL CONTROL STEP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,840	205.6(AFT)	0.8(LT)	14,840	6.0	290	LEVEL	ON	78

NOTE: 8-HELLFIRE CONFIGURATION

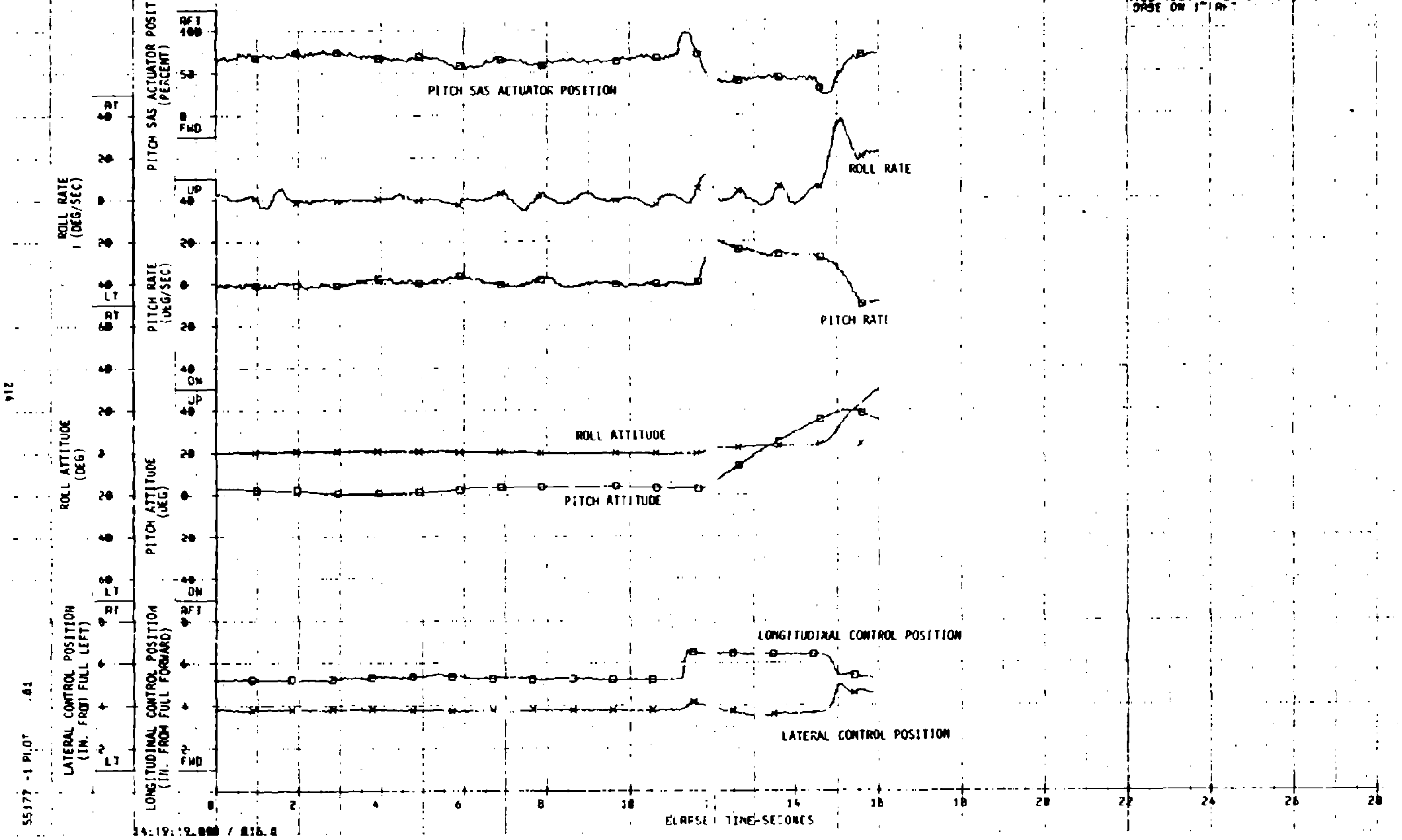

FIGURE 72

AFT LONGITUDINAL CONTROL STEP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,740	205.6(AFT)	0.8(LT)	14,860	5.5	290	LEVEL	OFF	78

NOTE: 8-HELLFIRE CONFIGURATION

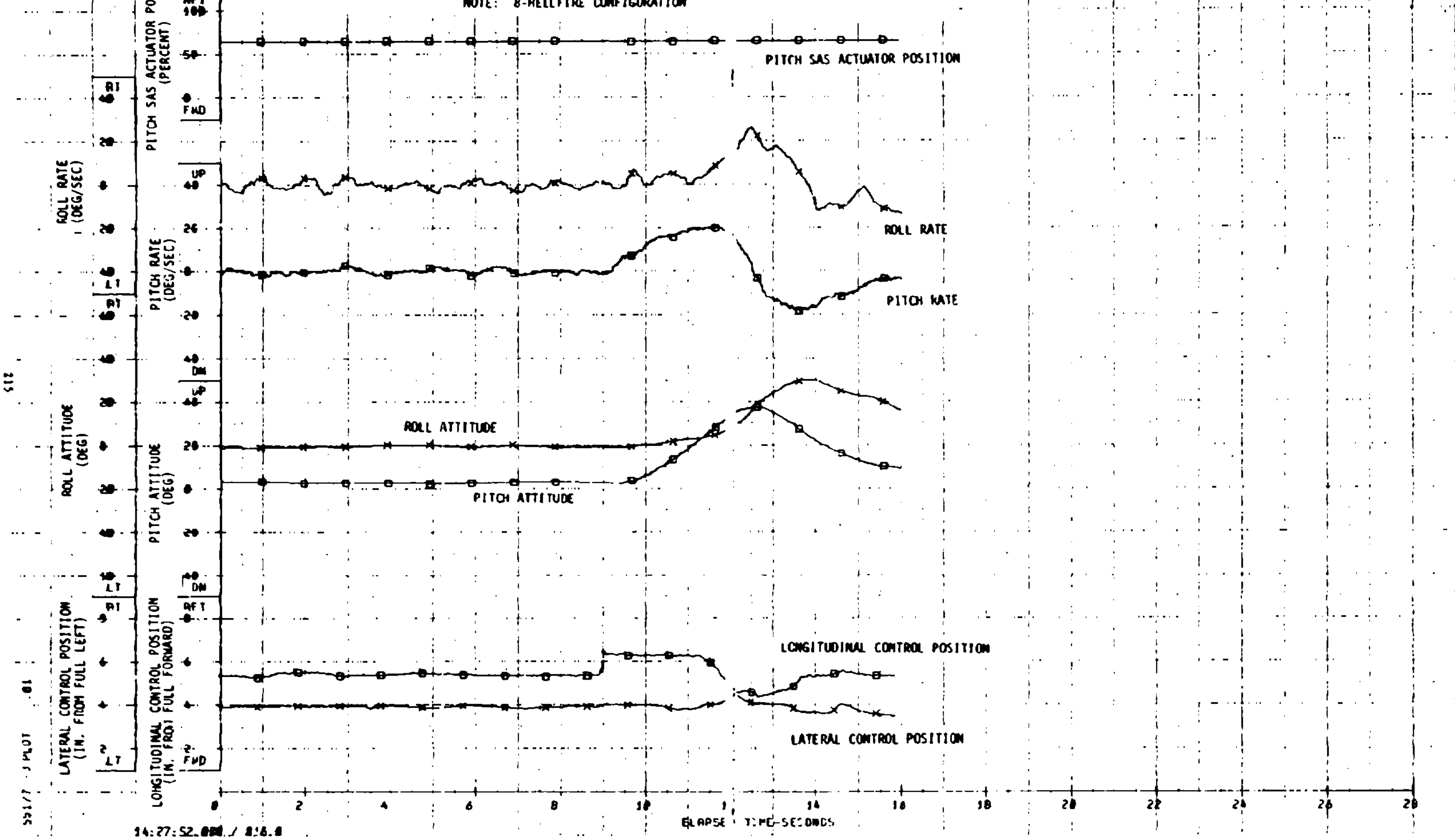

FIGURE 73
LATERAL CONTROLLABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	DASE CONDITION
○	15,140	205.9(AFT)	0.8(LT)	14,620	5.5	290	80	LEVEL	ON
□	14,760	205.9(AFT)	0.8(LT)	14,280	5.0	290	78	LEVEL	OFF

NOTE: 8-HELLFIRE CONFIGURATION

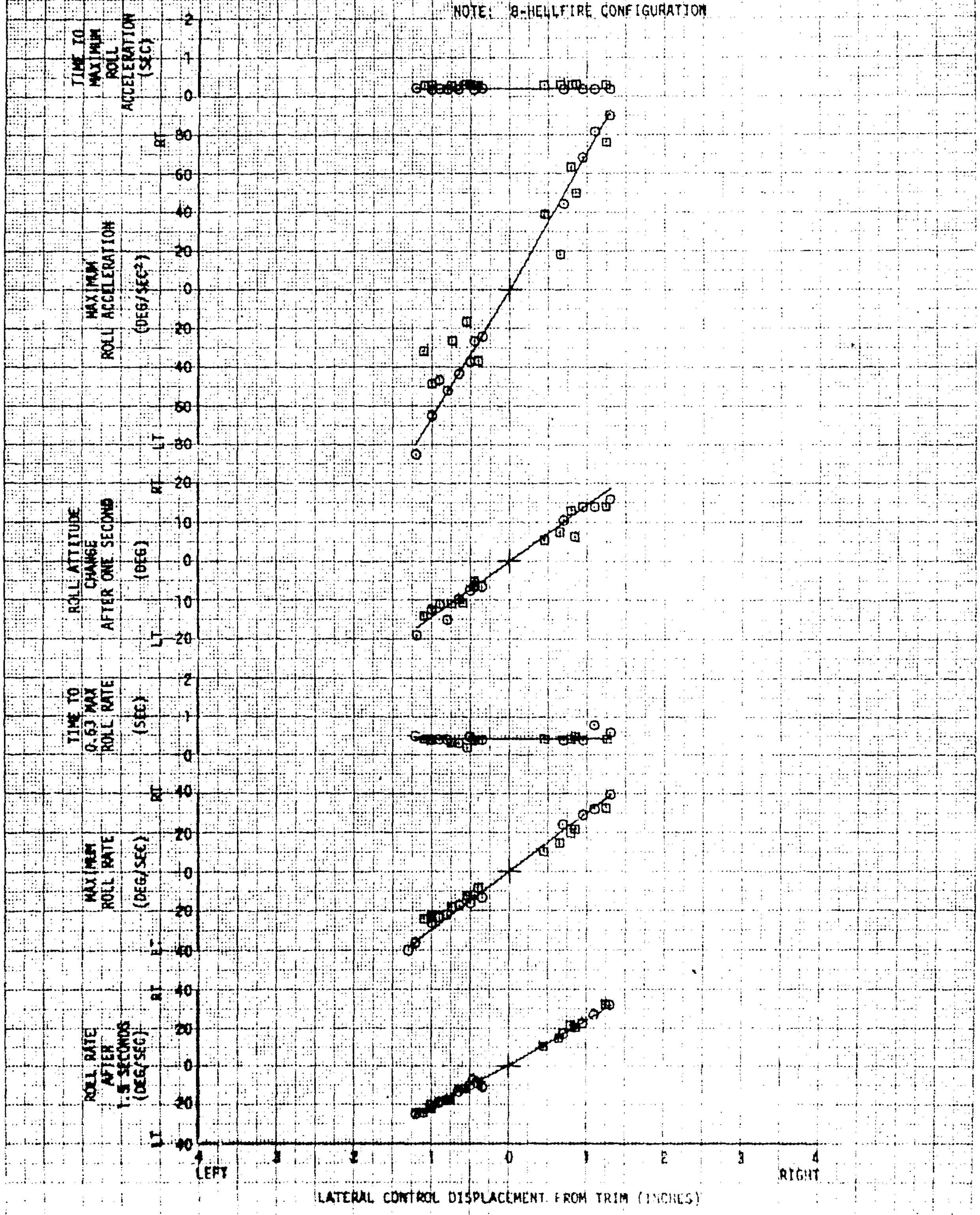

FIGURE 74

LEFT LATERAL CONTROL STEP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,120	205.6(AFT)	0.8(LT)	14,900	7.5	290	LEVEL	ON	78

NOTE: 8-HELLFIRE CONFIGURATION

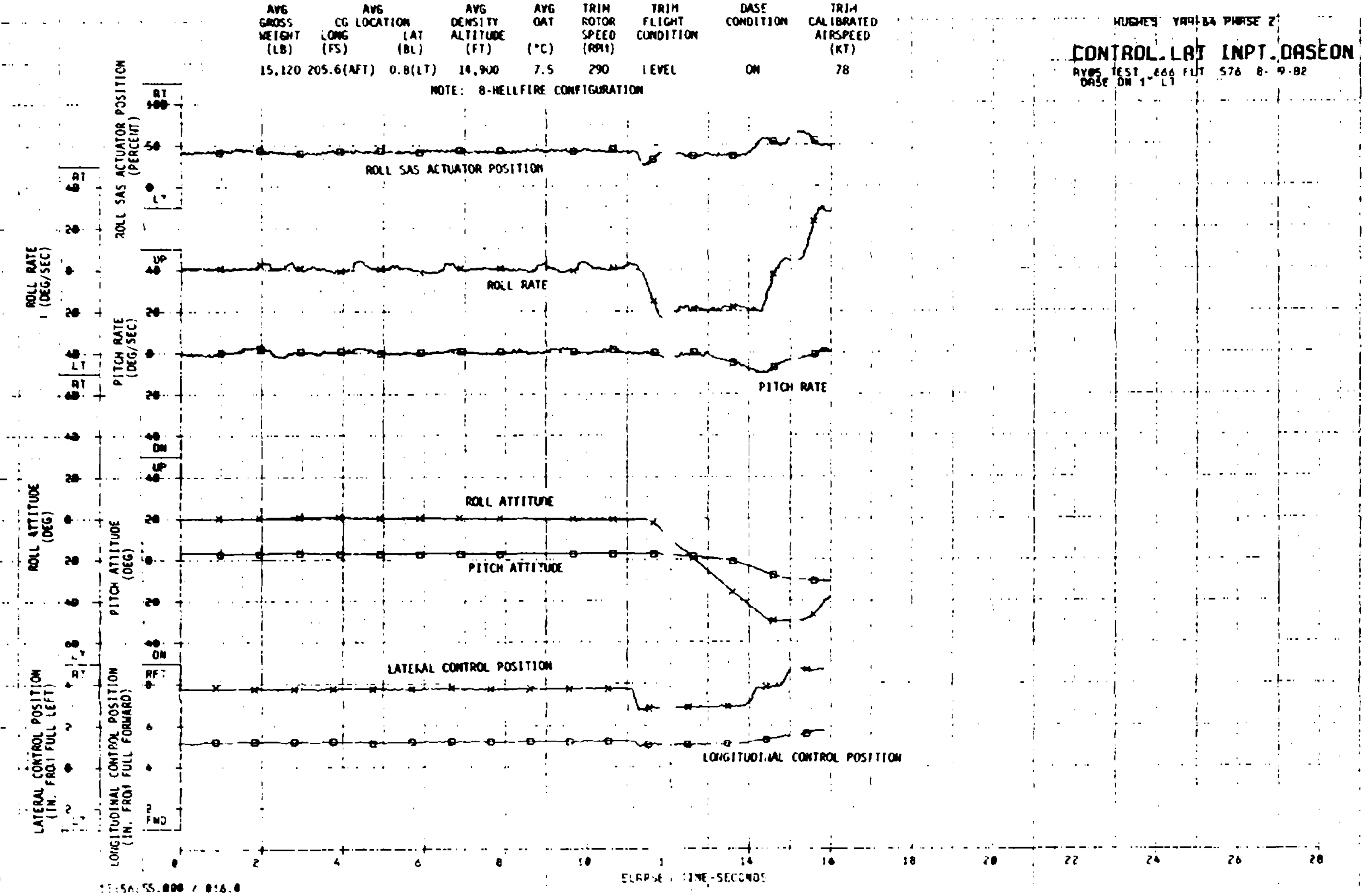

FIGURE 75

LEFT LATERAL CONTROL STEP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,620	295.5(AFT)	0.8(LT)	14,460	6.5	290	LEVEL	OFF	78

NOTE: 8-HELLFIRE CONFIGURATION

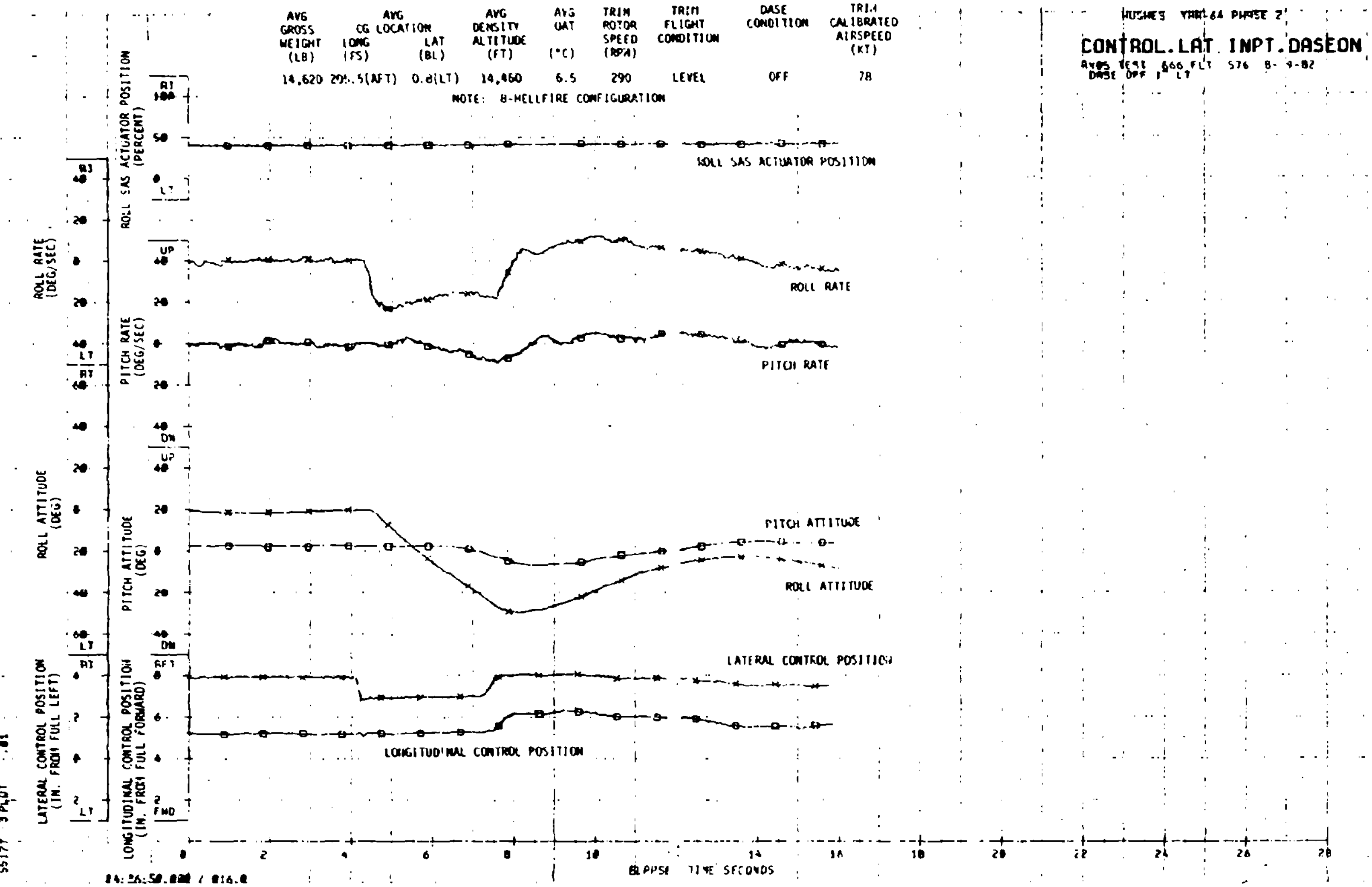

FIGURE 76
SLOPE LANDING - LEFT WHEEL UP

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	TRIM ROTOR SPEED (RPM)	DASE CONDITION
14,120	204.1(MID)	0.8(LT)	2760	29.5	289	ON

NOTES: 1. CLEAN CONFIGURATION WITH OUTBOARD PYLONS INSTALLED FOR STORES CLEARANCE DETERMINATION
2. 10.5° SLOPE ANGLE

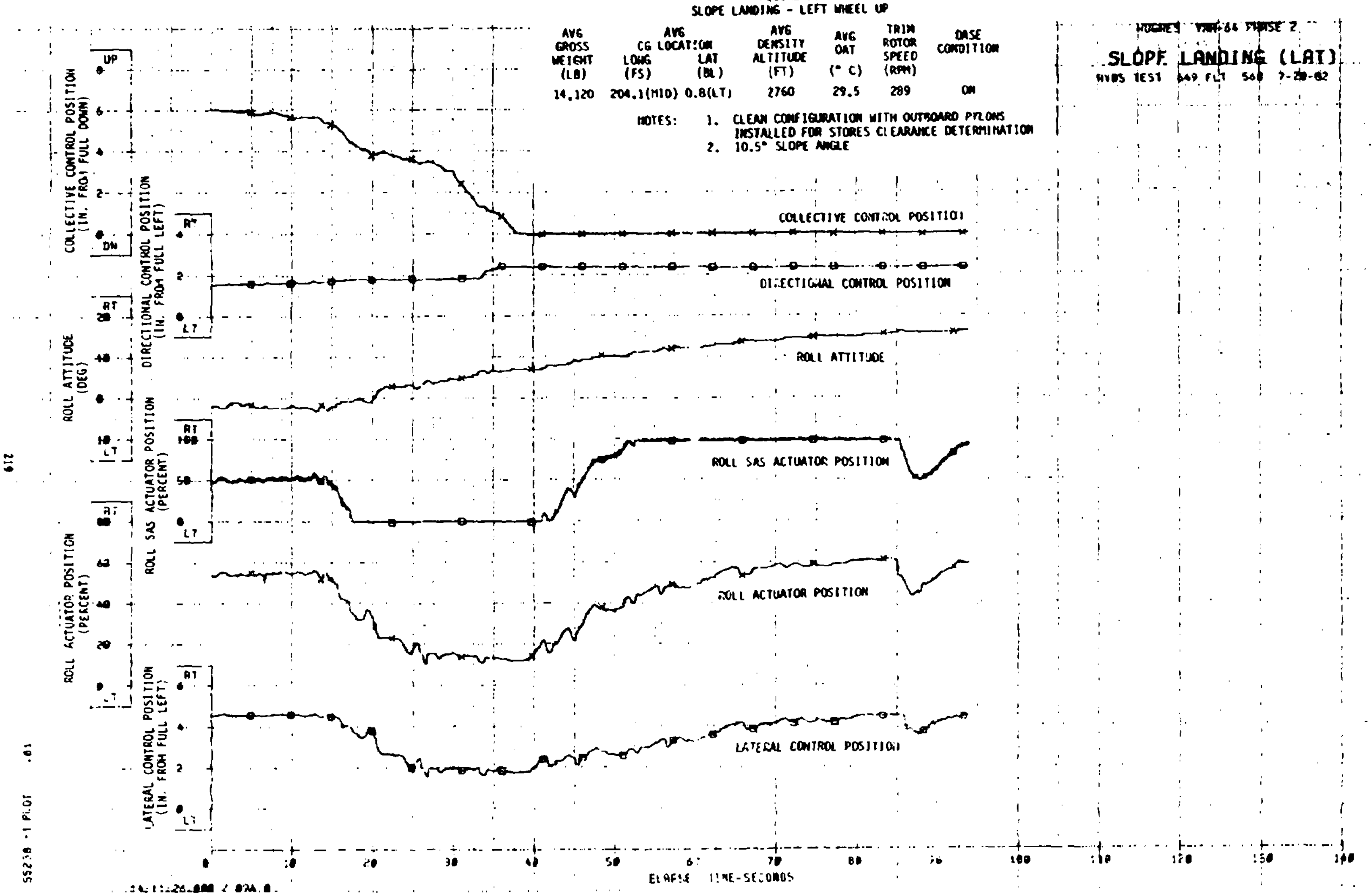

FIGURE 77
SLOPE LANDING - RIGHT WHEEL UP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM FLIGHT CONDITION	DASE CONDITION
14,420	205.4(AFT)	0.8(LT)	3120	33.5	290	ON

NOTES: 1. CLEAN CONFIGURATION WITH OUTBOARD PYLONS INSTALLED FOR STORES CLEARANCE DETERMINATION
2. 9° SLOPE ANGLE

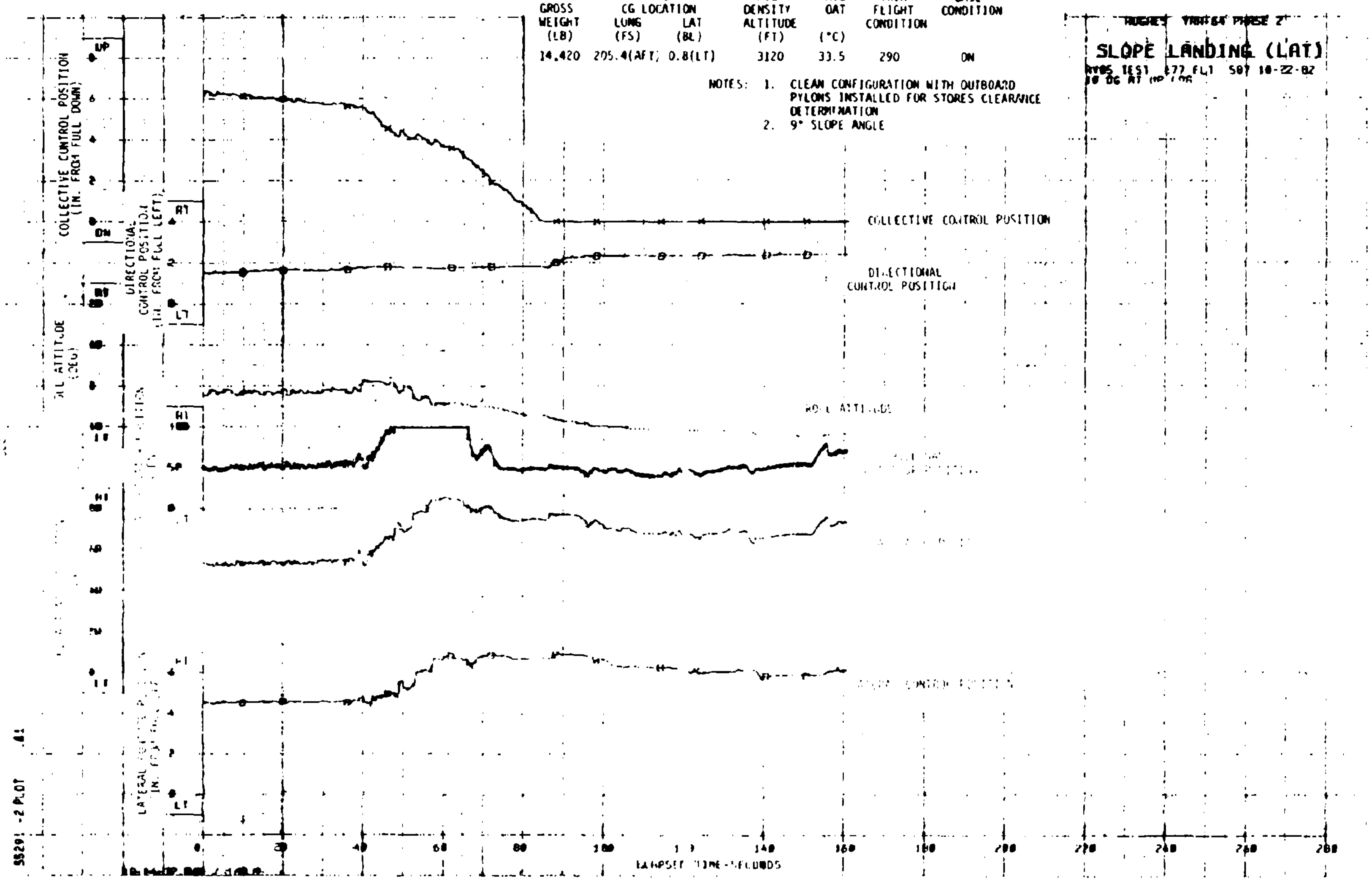

FIGURE 73

SLOPE LANDING NOSE UP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	BASE CONDITION
1?,800	204.1(MID)	0.0(LT)	2960	31.0	289	OH

NOTES: 1. CLEAN CONFIGURATION WITH OUTBOARD PYLONS INSTALLED FOR STORES CLEARANCE DETERMINATION
2. 12° SLOPE ANGLE

HUGHES YAH-64 PHASE 2
SLOPE LANDING (LONG.)
AVMS TEST 645 FLT 56B 7-20-82
12DEG NOSE UP LANDING

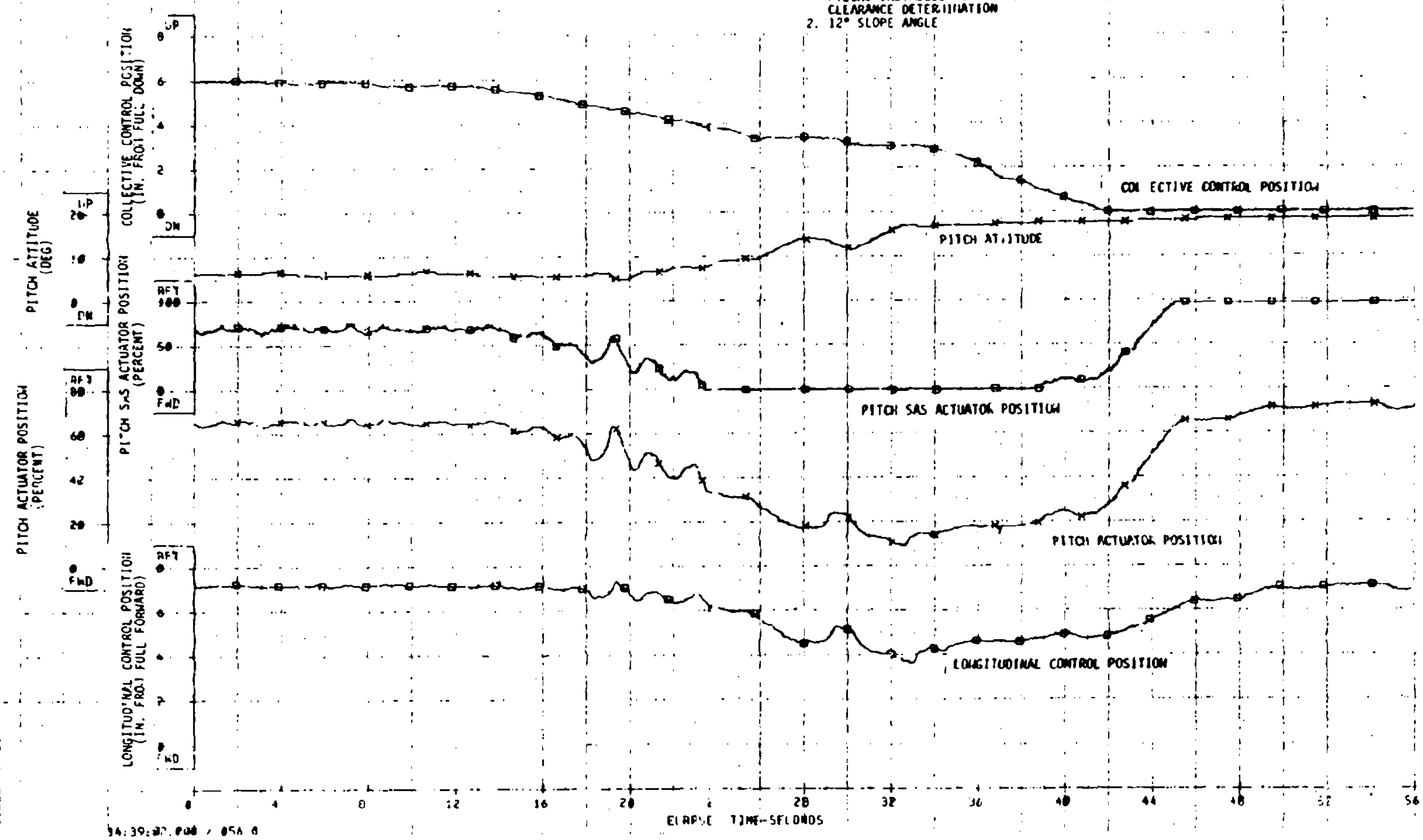

FIGURE 79
SLOPE LANDING - NOSE DOWN
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM FLIGHT CONDITION	DASE CONDITION
14,180	205.4(AFT)	0.8(LT)	3150	34.0	290	ON

NOTES: 1. CLEAN CONFIGURATION WITH OUTBOARD PYLONS INSTALLED FOR STORES CLEARANCE DETERMINATION
2. 10° SLOPE ANGLE

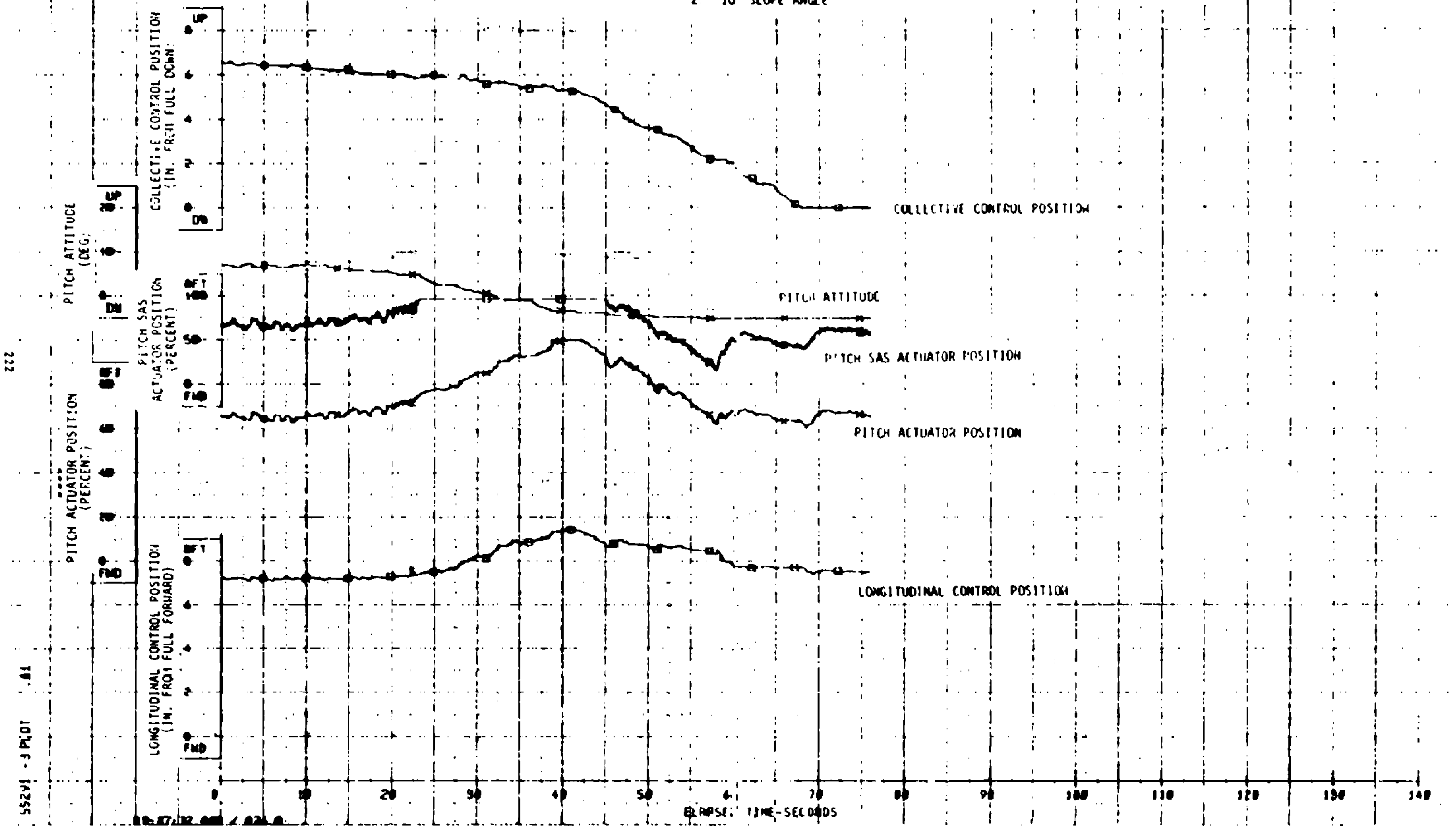

FIGURE 80
SIMULATED SINGLE ENGINE FAILURE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
15,300	205.6(AFT)	0.3(LT)	3700	35.0	291	OGE HOVER	ON

NOTE: 8-HELLFIRE CONFIGURATION

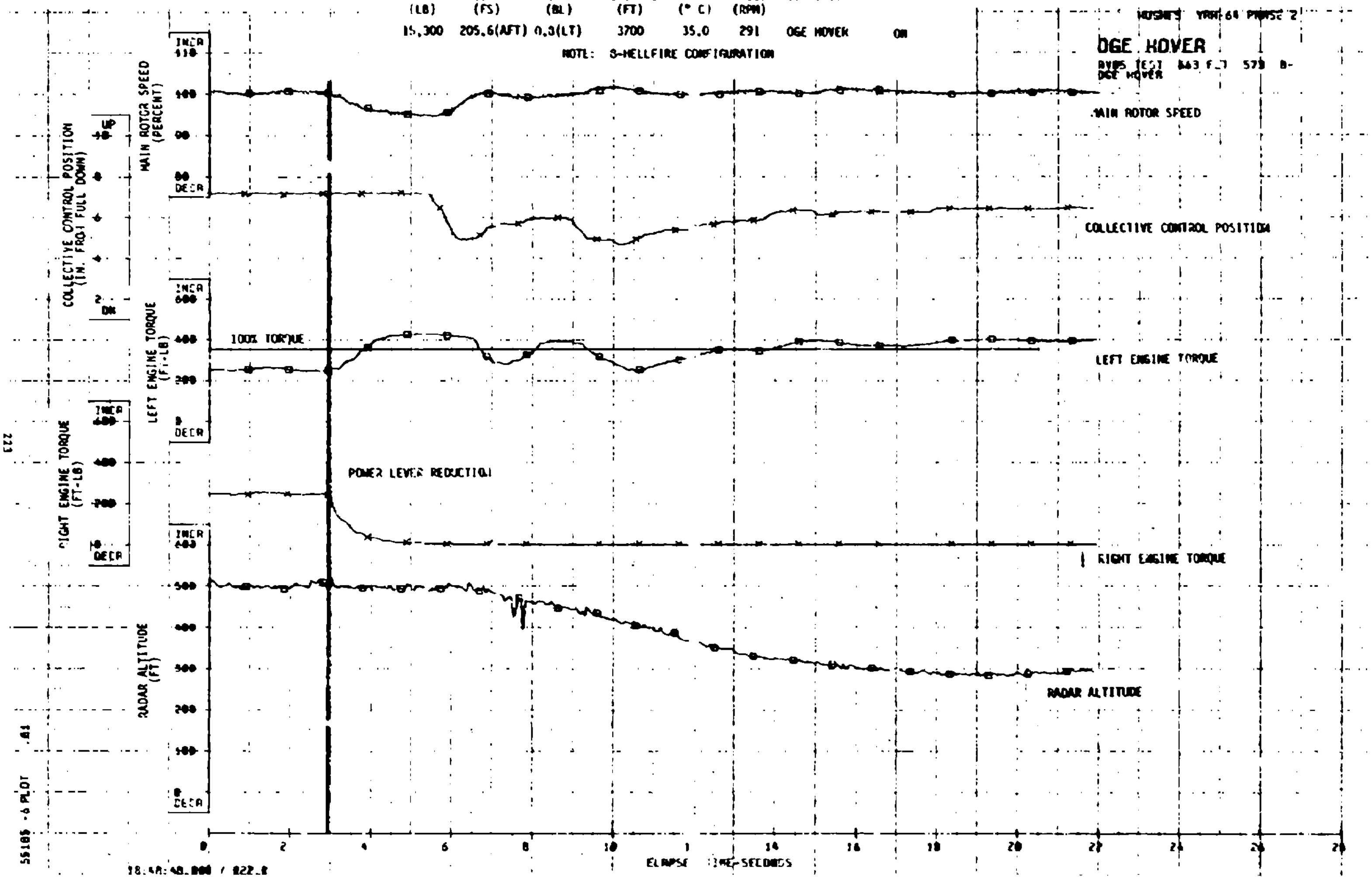

FIGURE 81
ENGINE RESPONSE TO COLLECTIVE RAMP INPUT (10-60% TORQUE)
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
16,000	204.0(MID)	0.8(LT)	7180	20.5	290	AUTOROTATION	ON	96

NOTE: 8-HELLFIRE CONFIGURATION

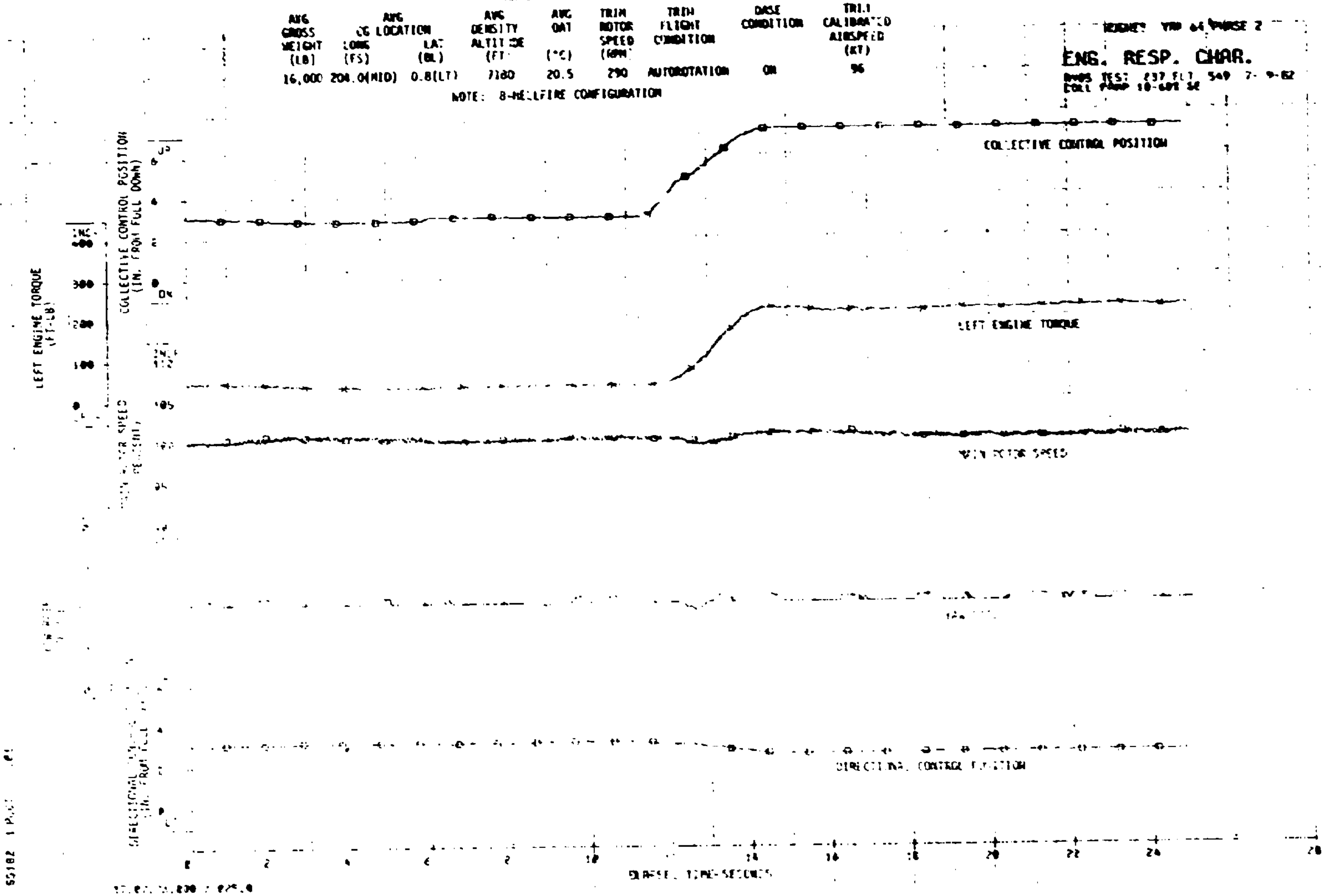

FIGURE 82

ENGINE RESPONSE TO COLLECTIVE RAMP INPUT (0-50% TORQUE)
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
13,480	204.0(MID)	0.8(LT)	5080	29.5	290	AUTOROTATION	ON	96

NOTE: CLEAN CONFIGURATION

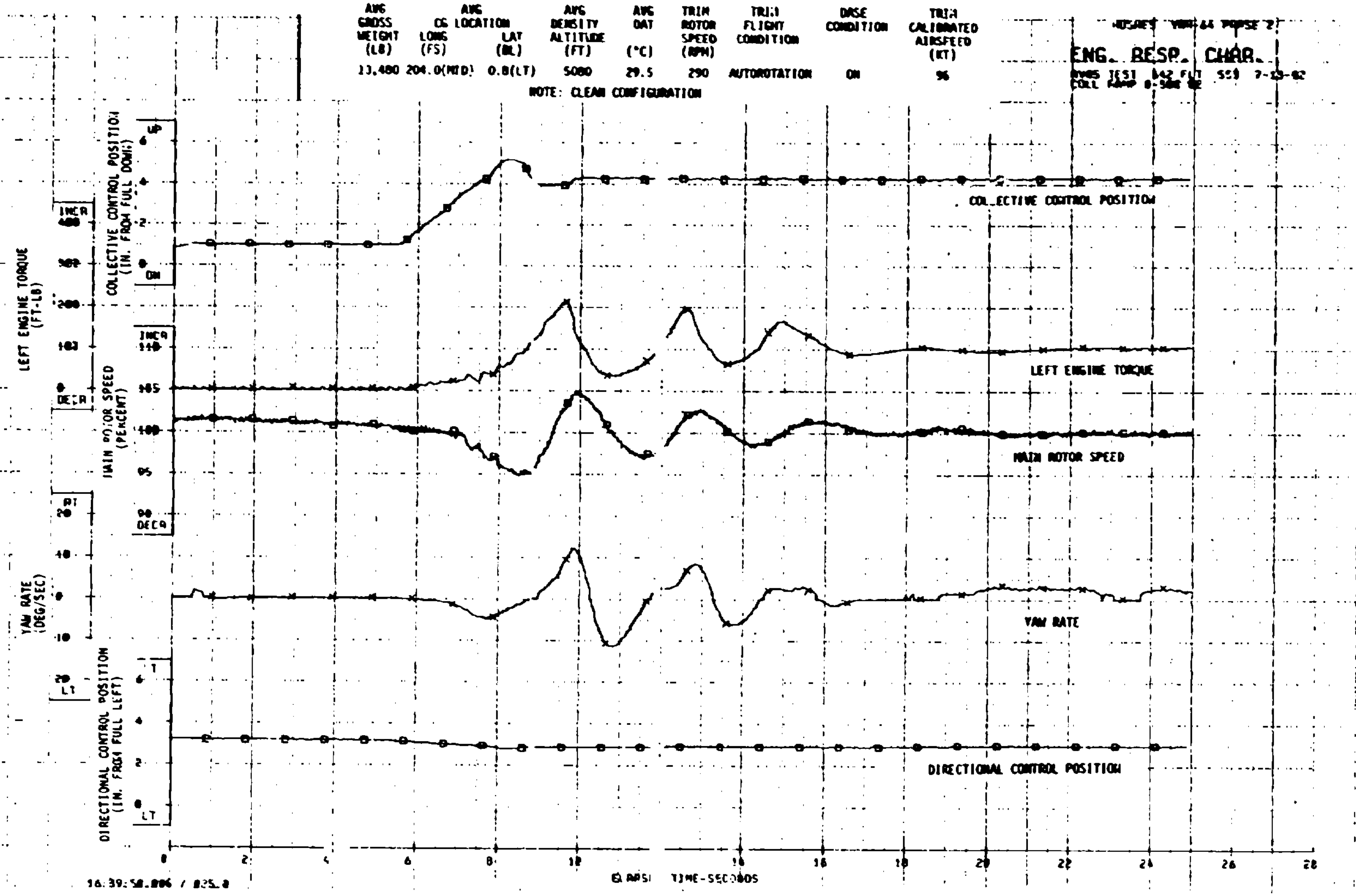

FIGURE 83

QUICK STOP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
13,880	205.8(AFT)	0.8(LT)	5840	23.0	290	LEVEL	ON	78

NOTE: CLEAN CONFIGURATION

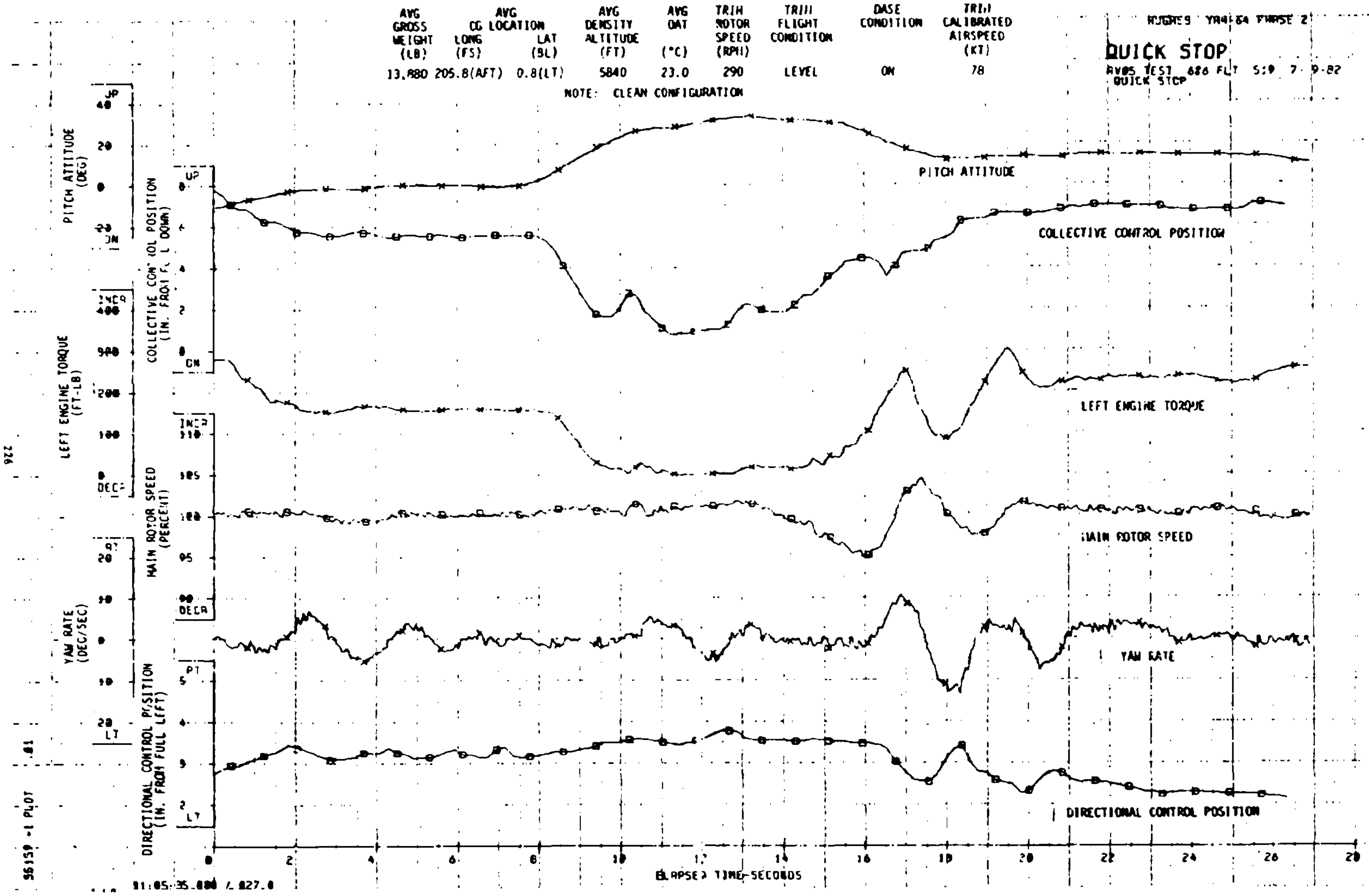

FIGURE 84

ENGINE RESPONSE TO GRADUAL AIRCRAFT DECELERATION
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,640	202.5(FWD)	0.8(LT)	5980	24.0	291	LEVEL	ON	96

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. COLLECTIVE CONTROL HELD FIXED

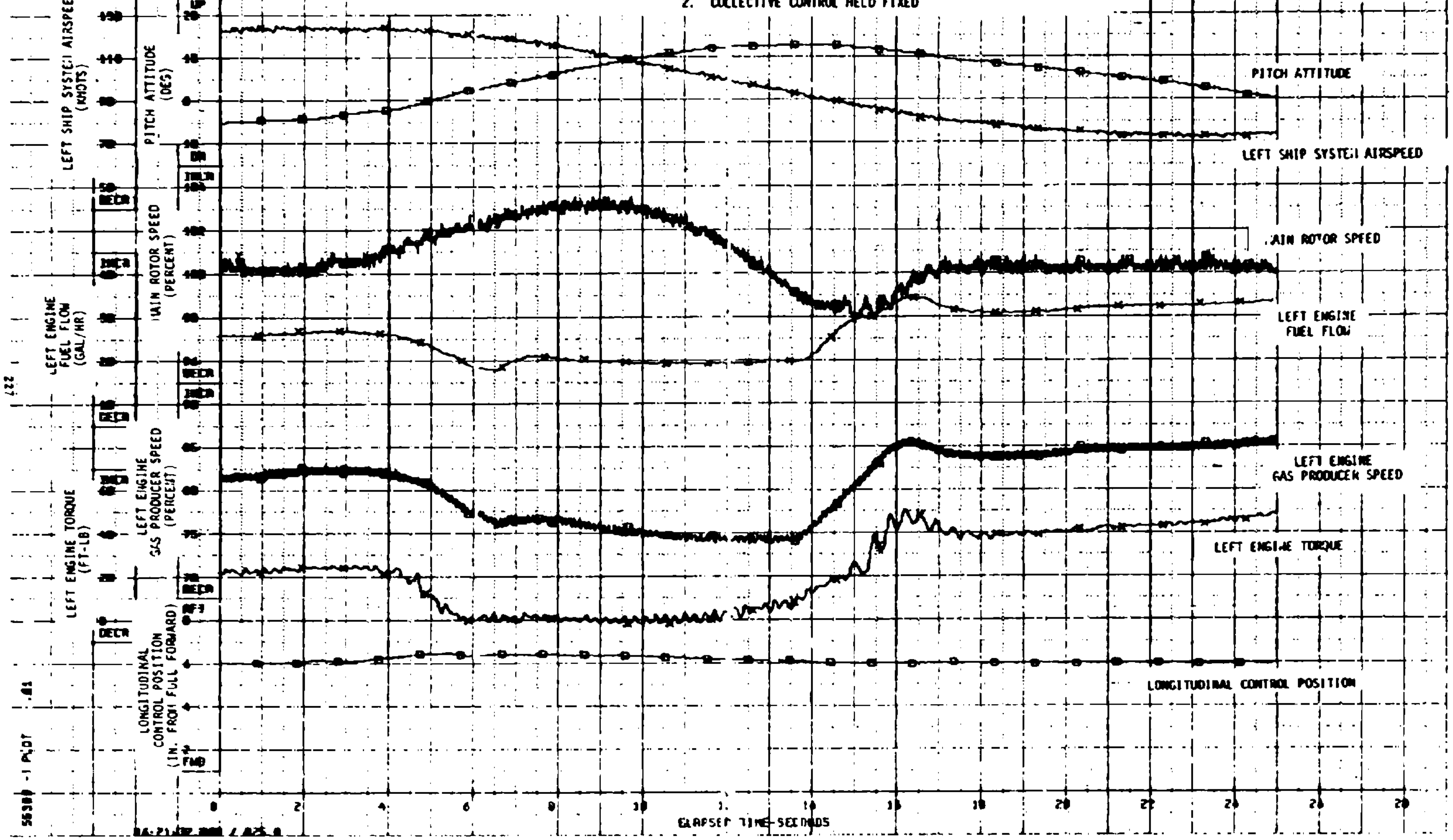

FIGURE 85

RETRIMMING WITH ATTITUDE HOLD ON
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,000	203.3(FWD)	0.8(LT)	10,360	17.5	290	LEVEL	ON	96

NOTE: 8-HELLFIRE CONFIGURATION

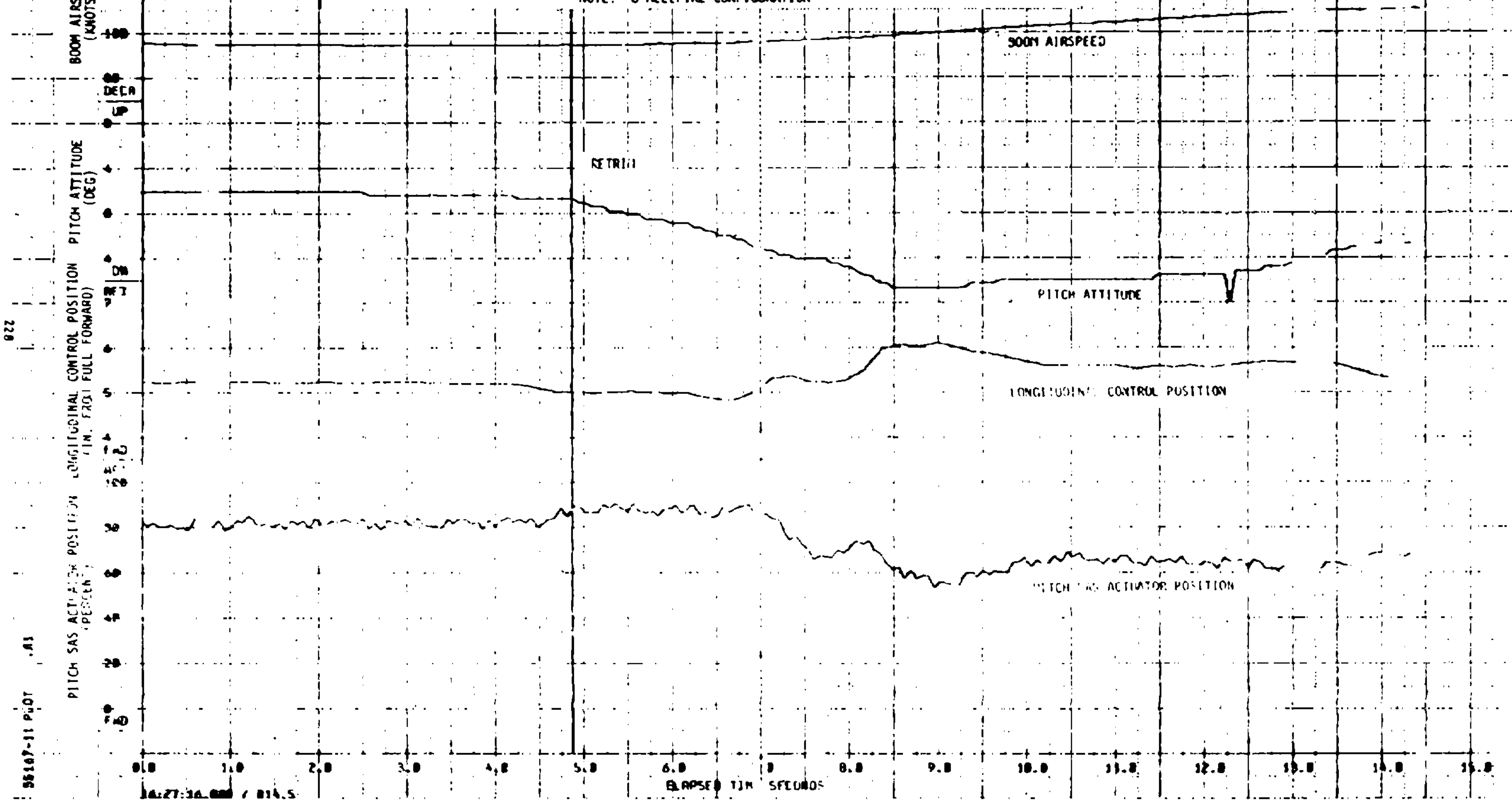

FIGURE 86

RETRIMMING WITH ATTITUDE HOLD ON
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,000	203.3(FWD)	0.8(LT)	9960	18.5	290	LEVEL	ON	1[illegible]

NOTE: 8-HELLFIRE CONFIGURATION

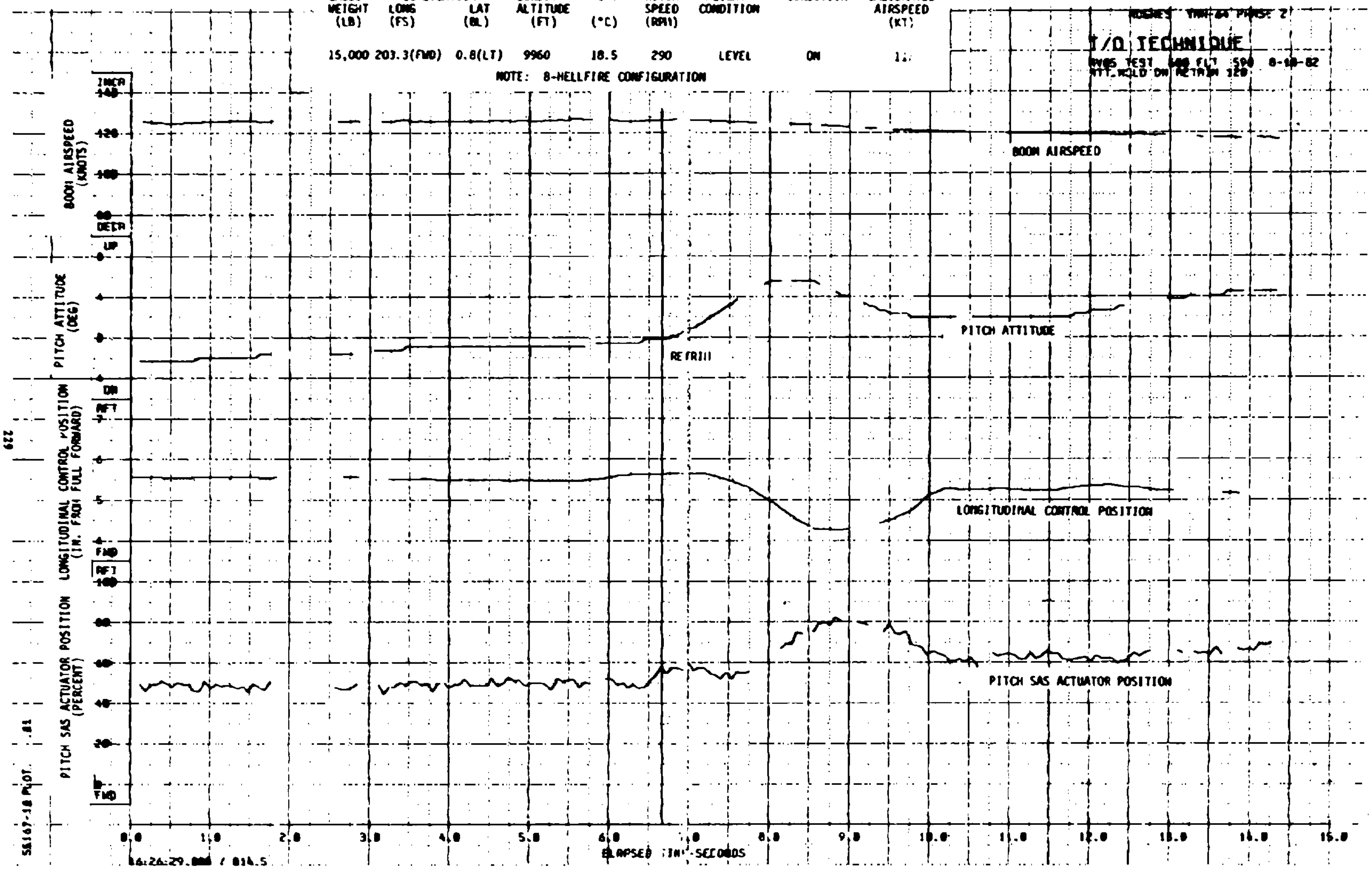

FIGURE 87

DASE DISENGAGEMENT - ATTITUDE HOLD ON
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,060	202.5(FWD)	0.8(LT)	6180	24.5	290	LEVEL	ON	96

NOTE: 8-HELLFIRE CONFIGURATION

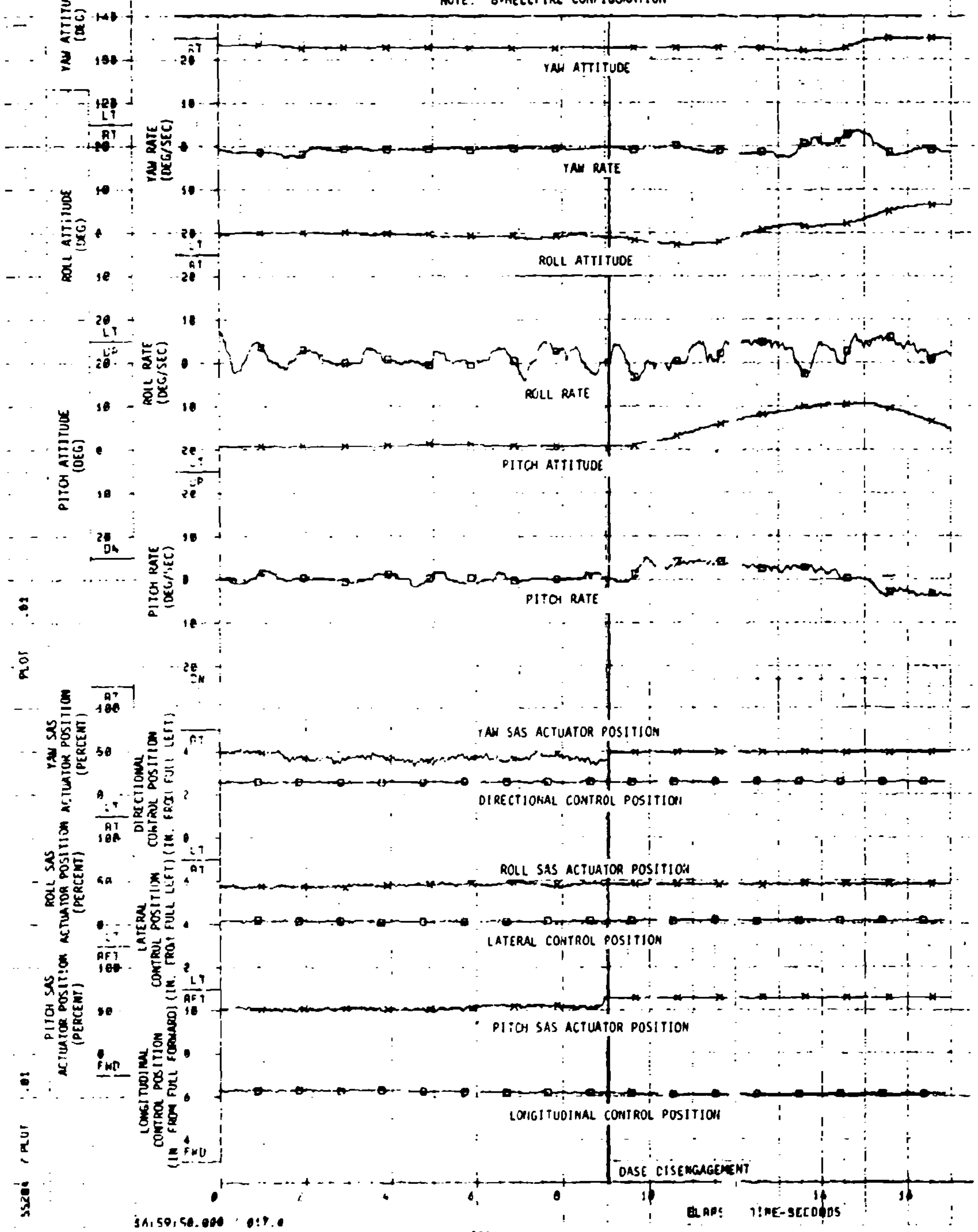

FIGURE 88

DASE DISENGAGEMENT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,100	202.5(FWD)	0.8(LT)	6600	25.0	290	LEVEL	ON	98

NOTE: 8-HELLFIRE CONFIGURATION

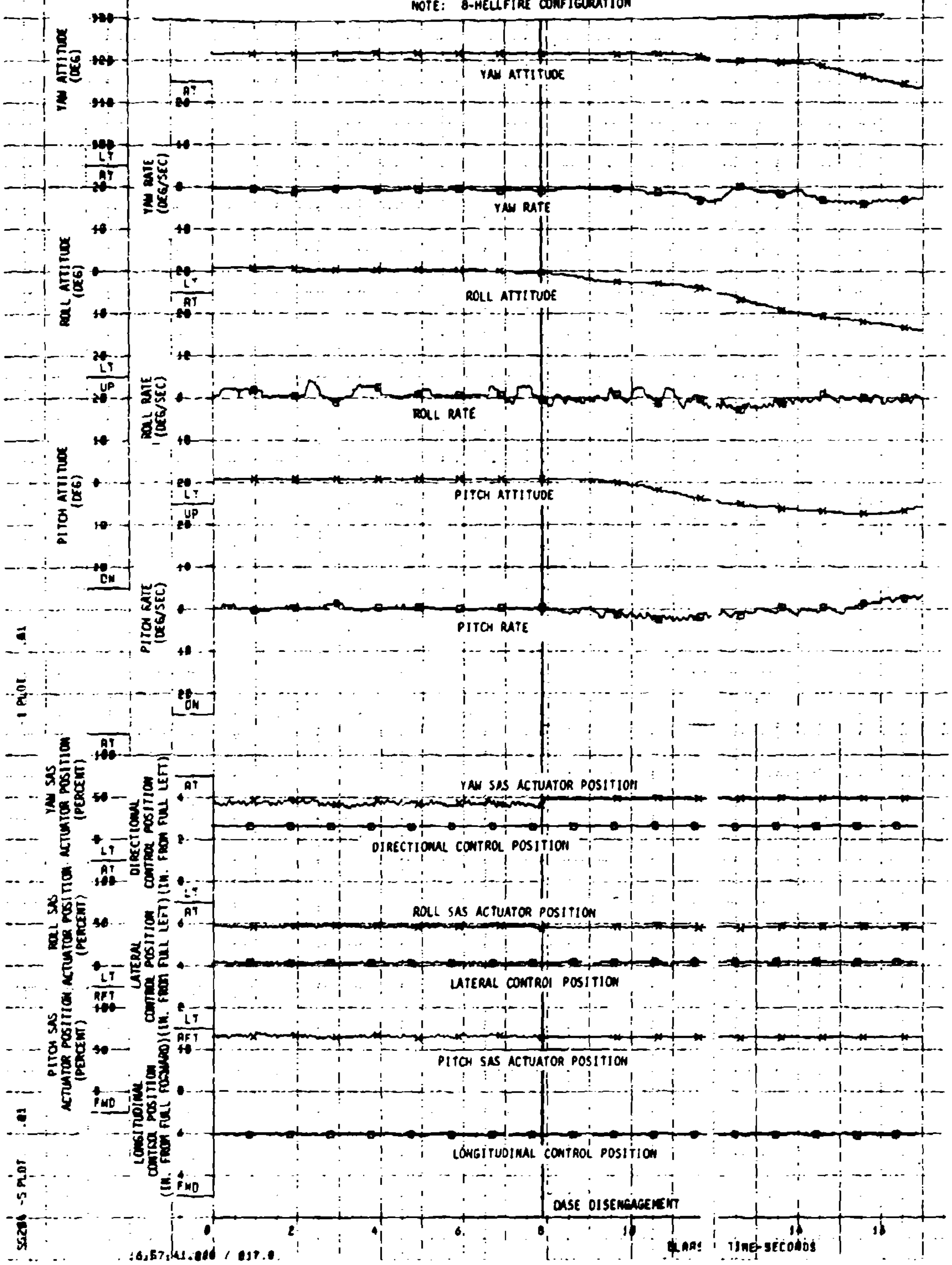

FIGURE 69

DASE DISENGAGEMENT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,140	202.5(FWD)	0.8(LT)	6680	24.0	290	LEVEL	ON	43

NOTE: 8-HELLFIRE CONFIGURATION

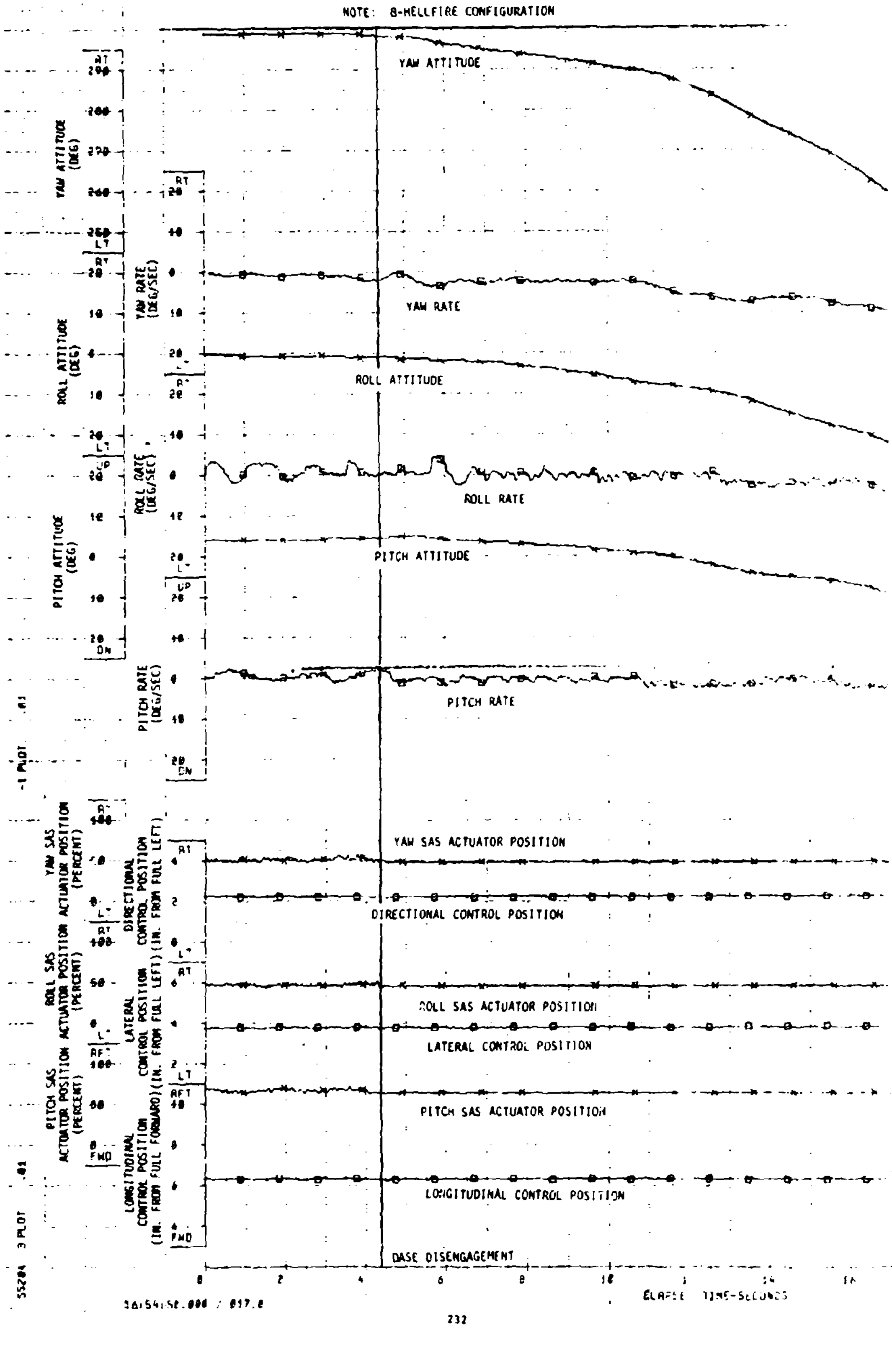

FIGURE 90

DASE DISENGAGEMENT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,980	202.5(FWD)	0.8(LT)	5820	27.0	290	LEVEL	ON	140

NOTE: 8-HELLFIRE CONFIGURATION

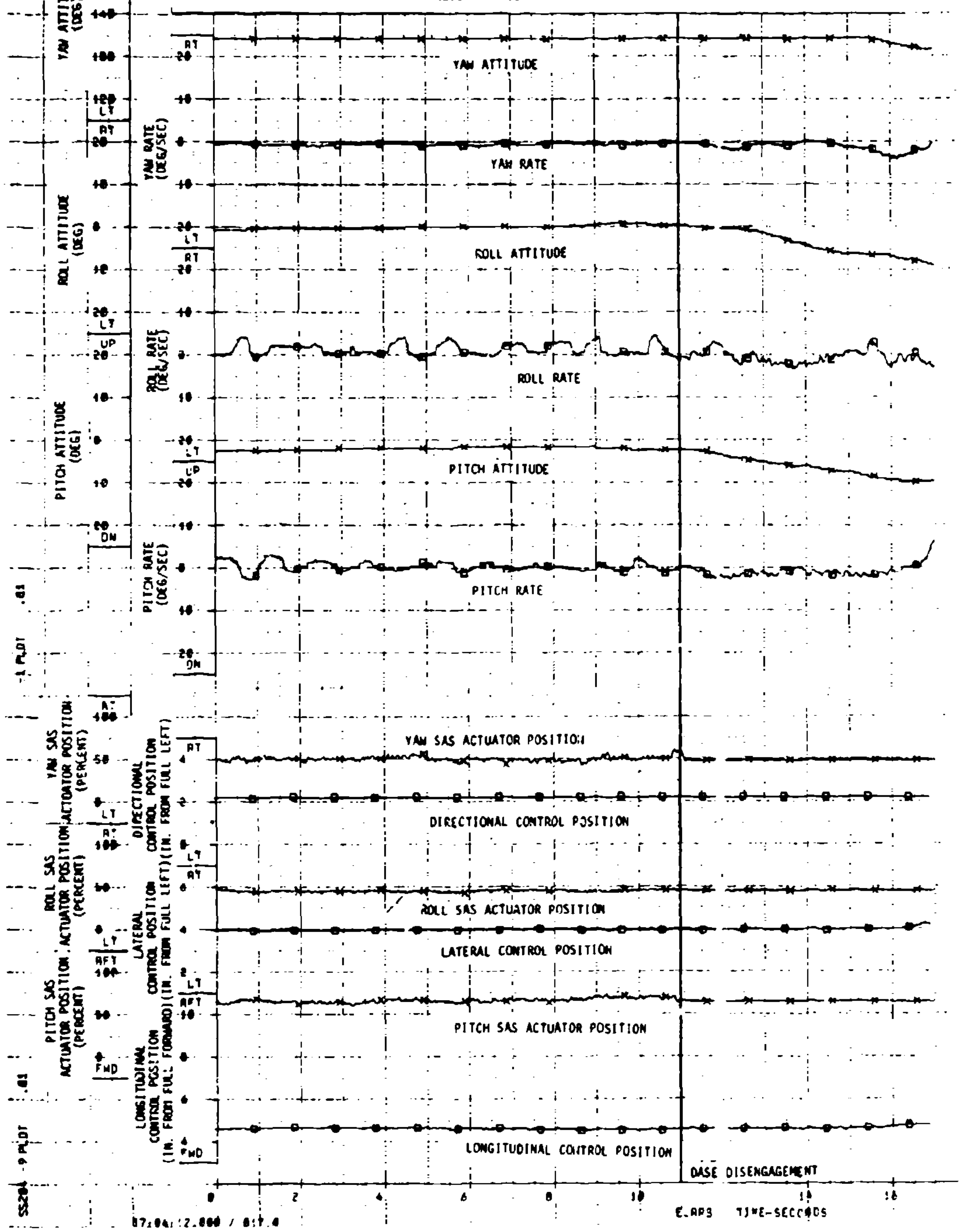

FIGURE 91
DASE DISENGAGEMENT - LOW-SPEED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	DASE CONDITION
15,400	203.1(FWD)	-0.8(LT)	2820	30.0	289	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. HOVER AUGMENTATION SYSTEM ON
3. AIRSPEED LESS THAN 15 KNOTS

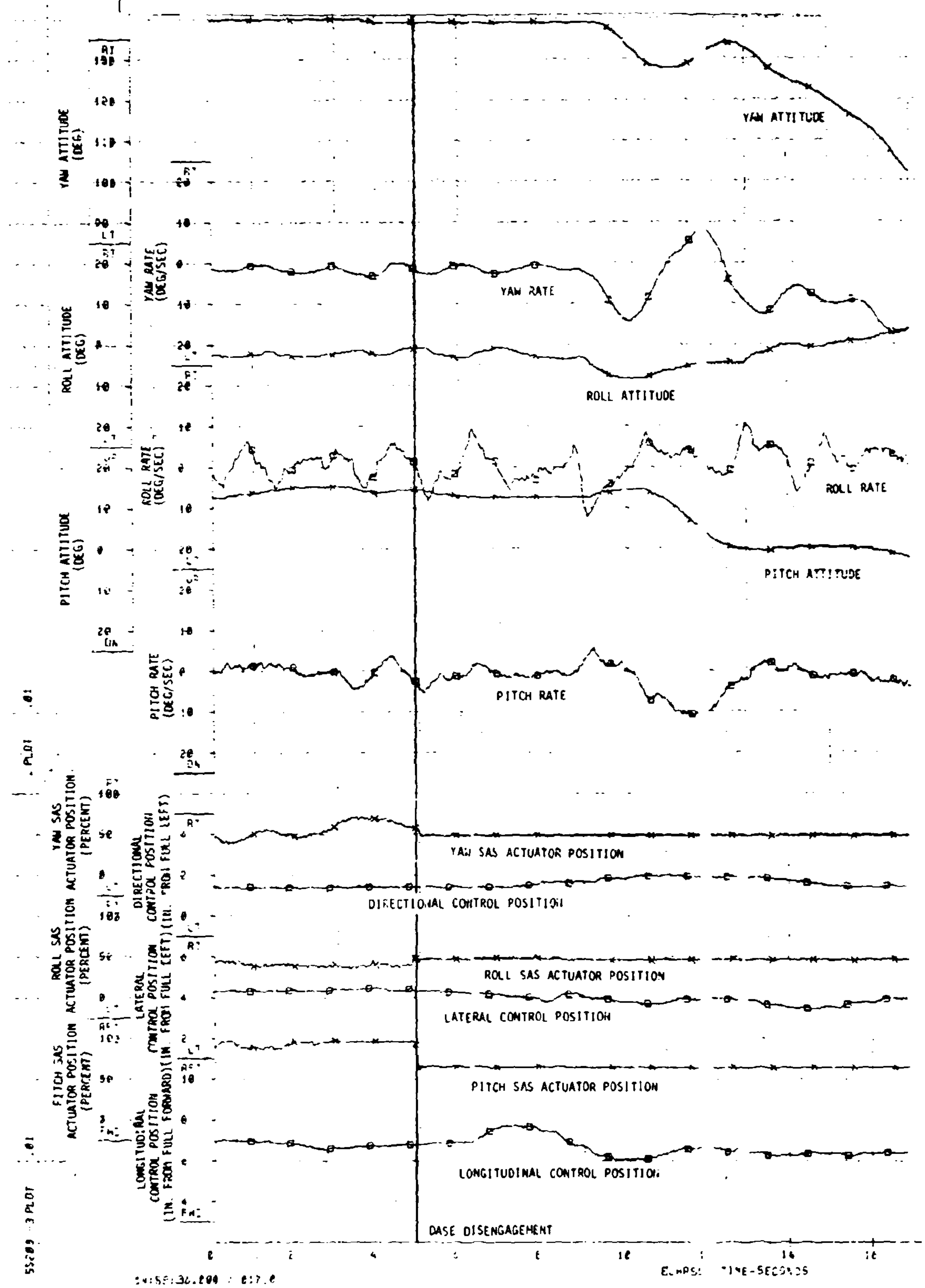

FIGURE 92
DASE DISENGAGEMENT - RIGHT SIDEWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	TRIM ROTOR SPEED (RPM)	DASE CONDITION
15,860	203.1(FWD)	0.8(LT)	2680	29.5	290	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. HOVER AUGMENTATION SYSTEM ON
3. AIRSPEED LESS THAN 15 KNOTS

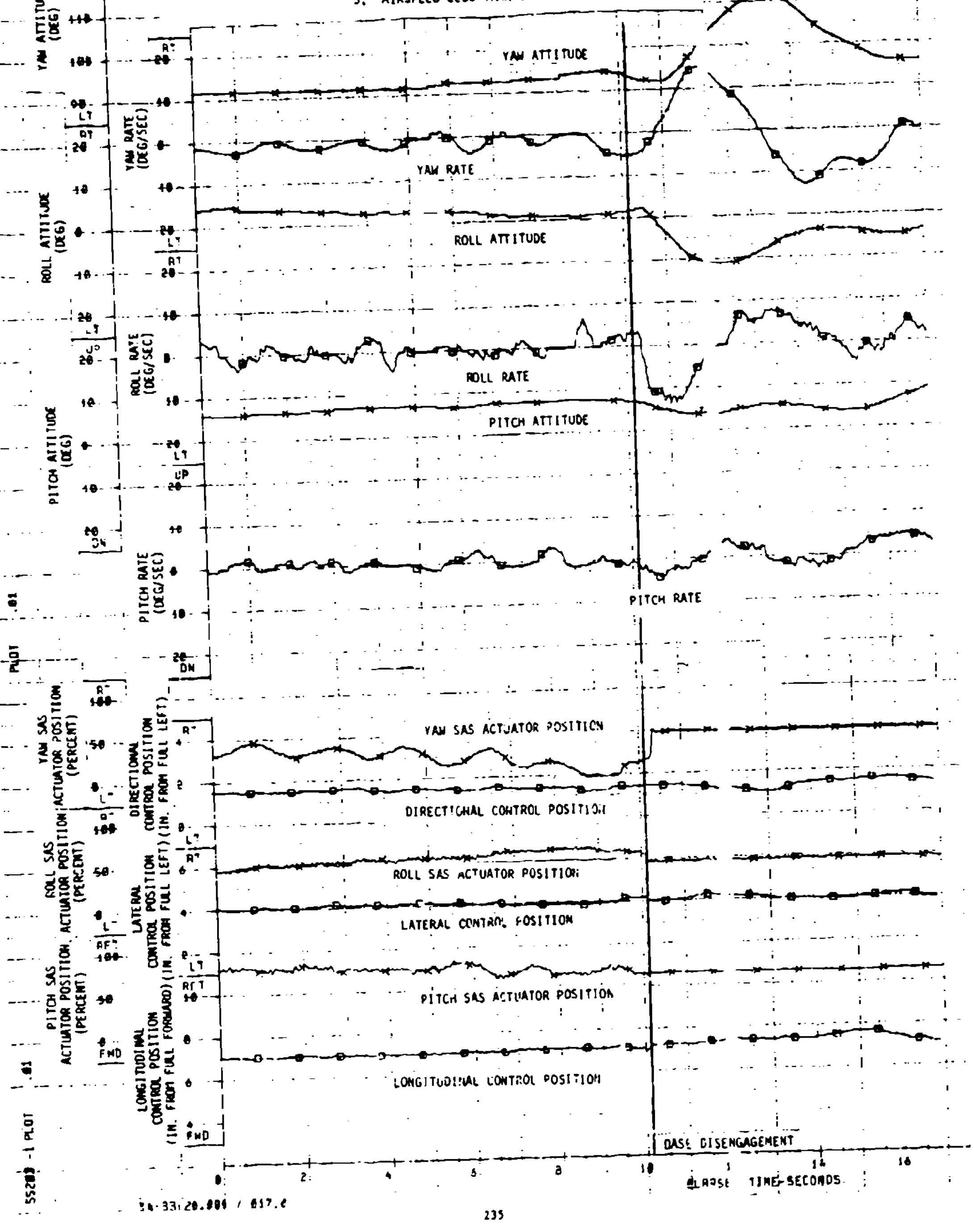

FIGURE 93

DASE DISENGAGEMENT - REARWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	DASE CONDITION
15,360	203.1(FWD)	0.8(LT)	2900	30.0	290	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. HOVER AUGMENTATION SYSTEM ON
3. AIRSPEED LESS THAN 15 KNOTS

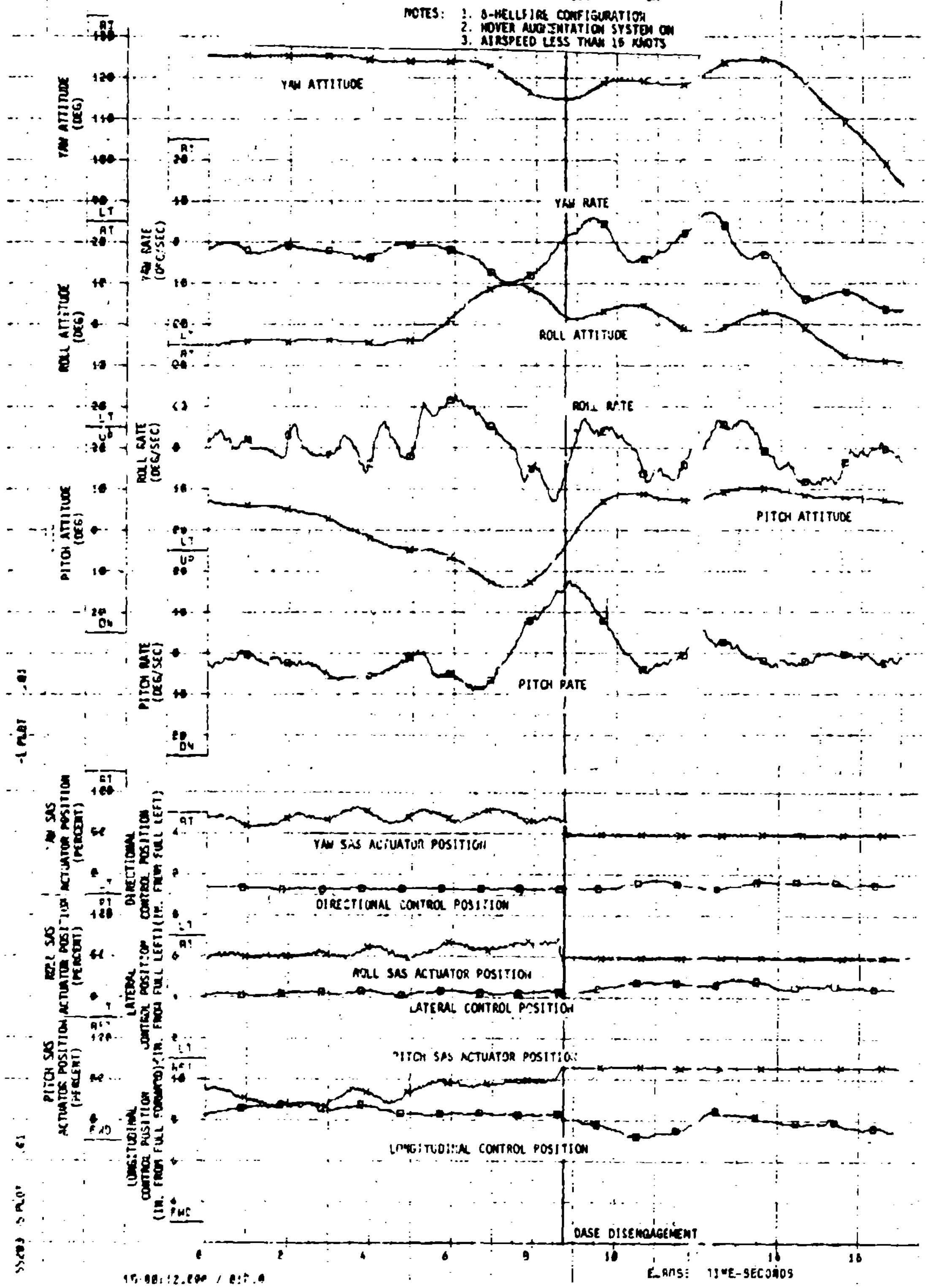

FIGURE 94

DASE DISENGAGEMENT - HOVER
YAH-64 USA S/N 77-23258

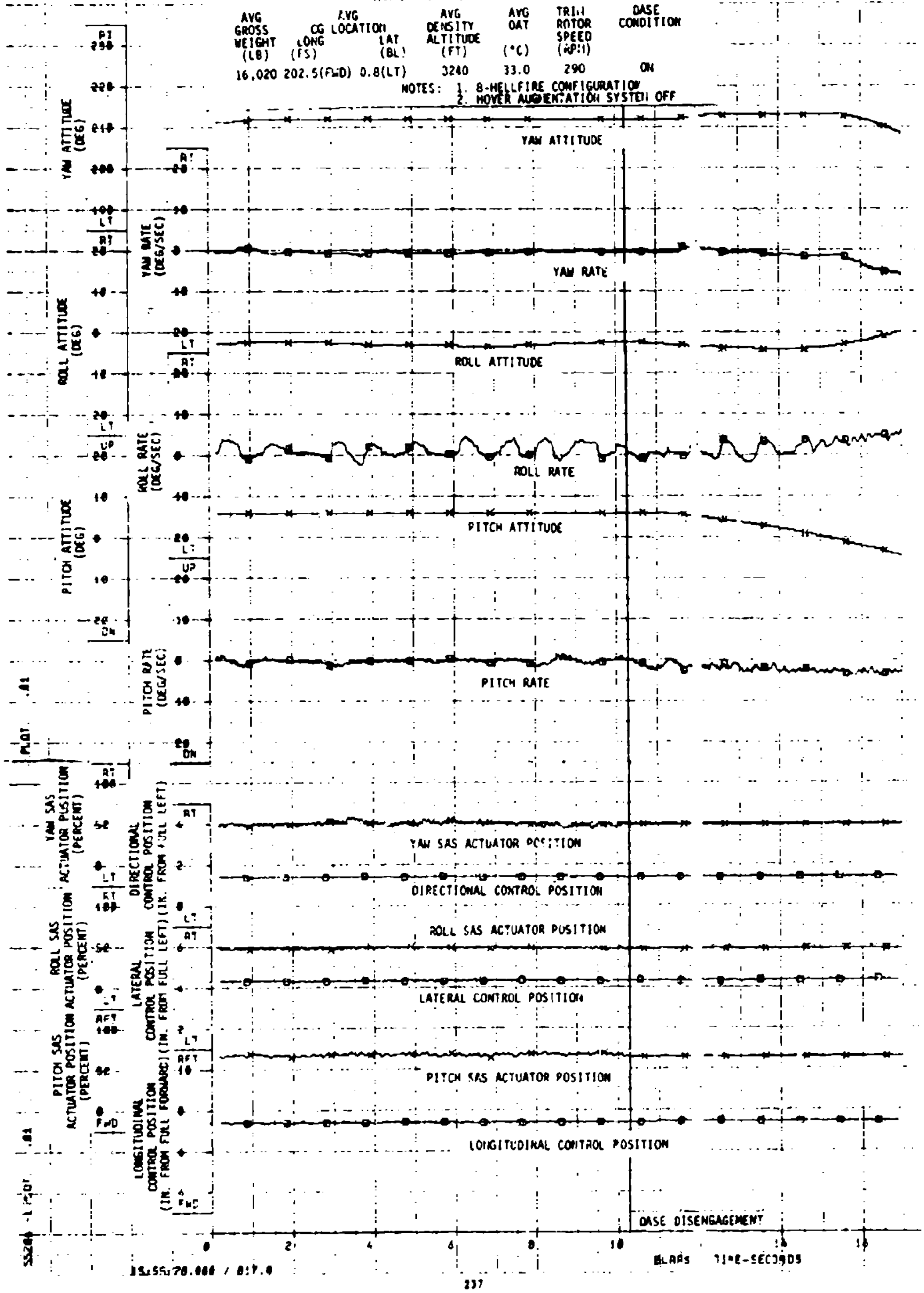

FIGURE 95
INSTRUMENT TAKEOFF
YAH-64 USA S/N 77-23256

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	SAS CONDITION
15,660	205.0(AFT)	0.8(LT)	4360	39.5	290	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. LIGHT TO MODERATE TURBULENCE

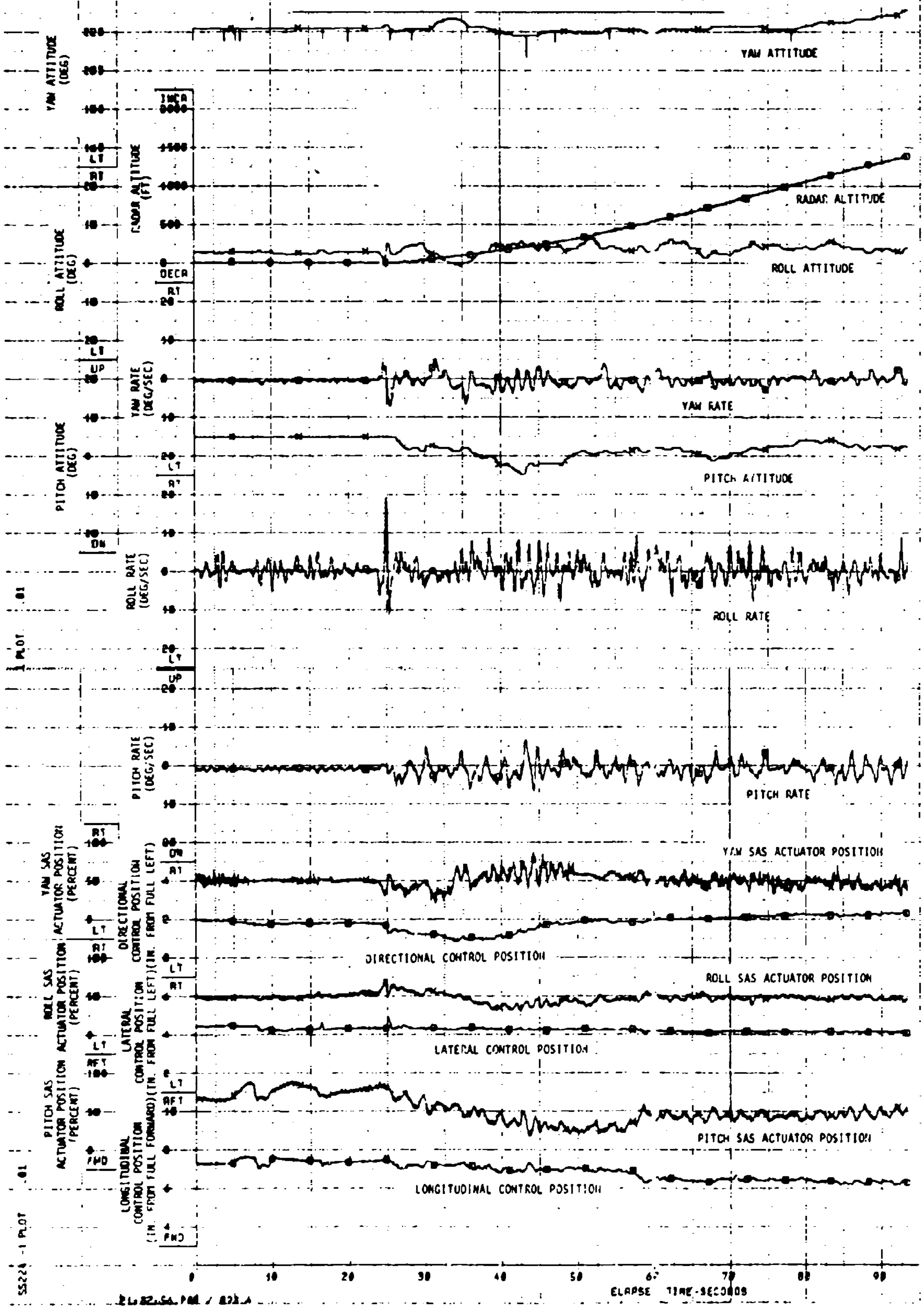

FIGURE 96

SIMULATED SINGLE-ENGINE FAILURE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,620	205.6(AFT)	0.8(LT)	7520	28.0	290	LEVEL	ON	134

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. PILOT REACTION 2 SECONDS AFTER SIMULATED FAILURE

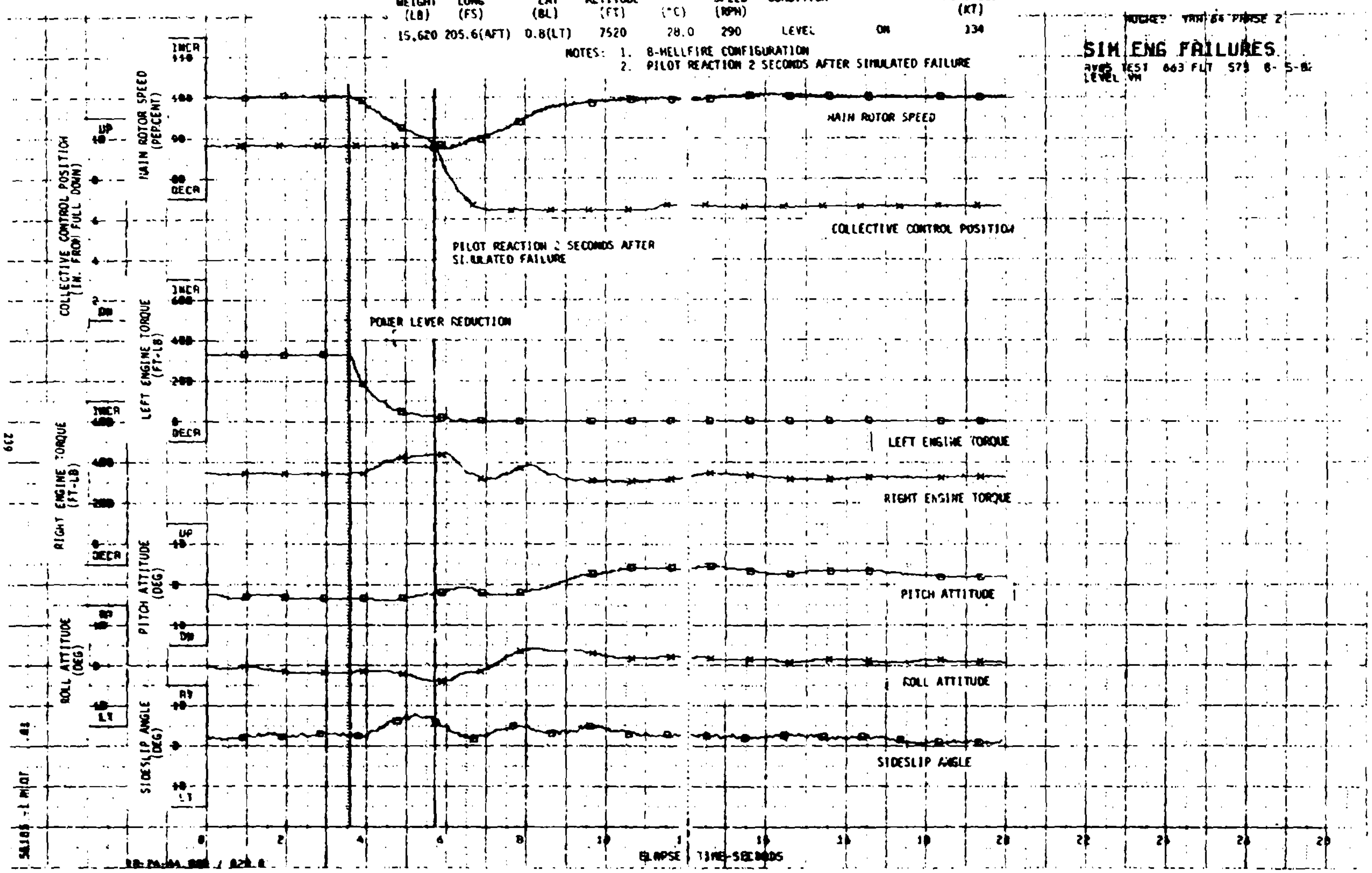

FIGURE 97
SIMULATED SINGLE-ENGINE FAILURE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
15,520	205.6(AFT)	0.8(LT)	7340	28.5	290	LEVEL	ON	135

NOTES: 1. 9-HELLFIRE CONFIGURATION
2. PILOT REACTION IMMEDIATELY AFTER LOW AUDIO WARNING

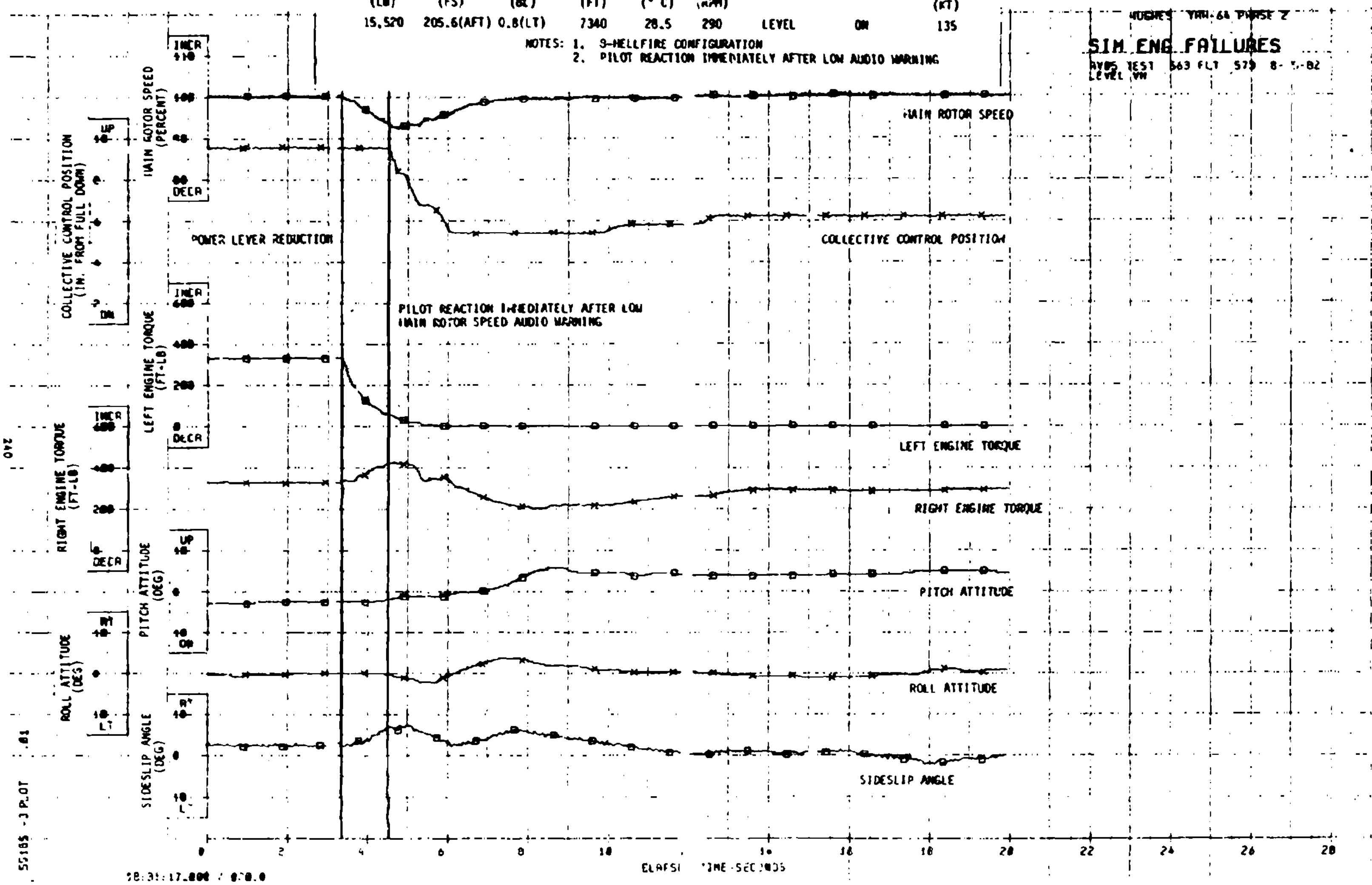

FIGURE 98
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,360	202.1(FWD)	0.8(LT)	15,800	3.5	290	LEVEL

NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION

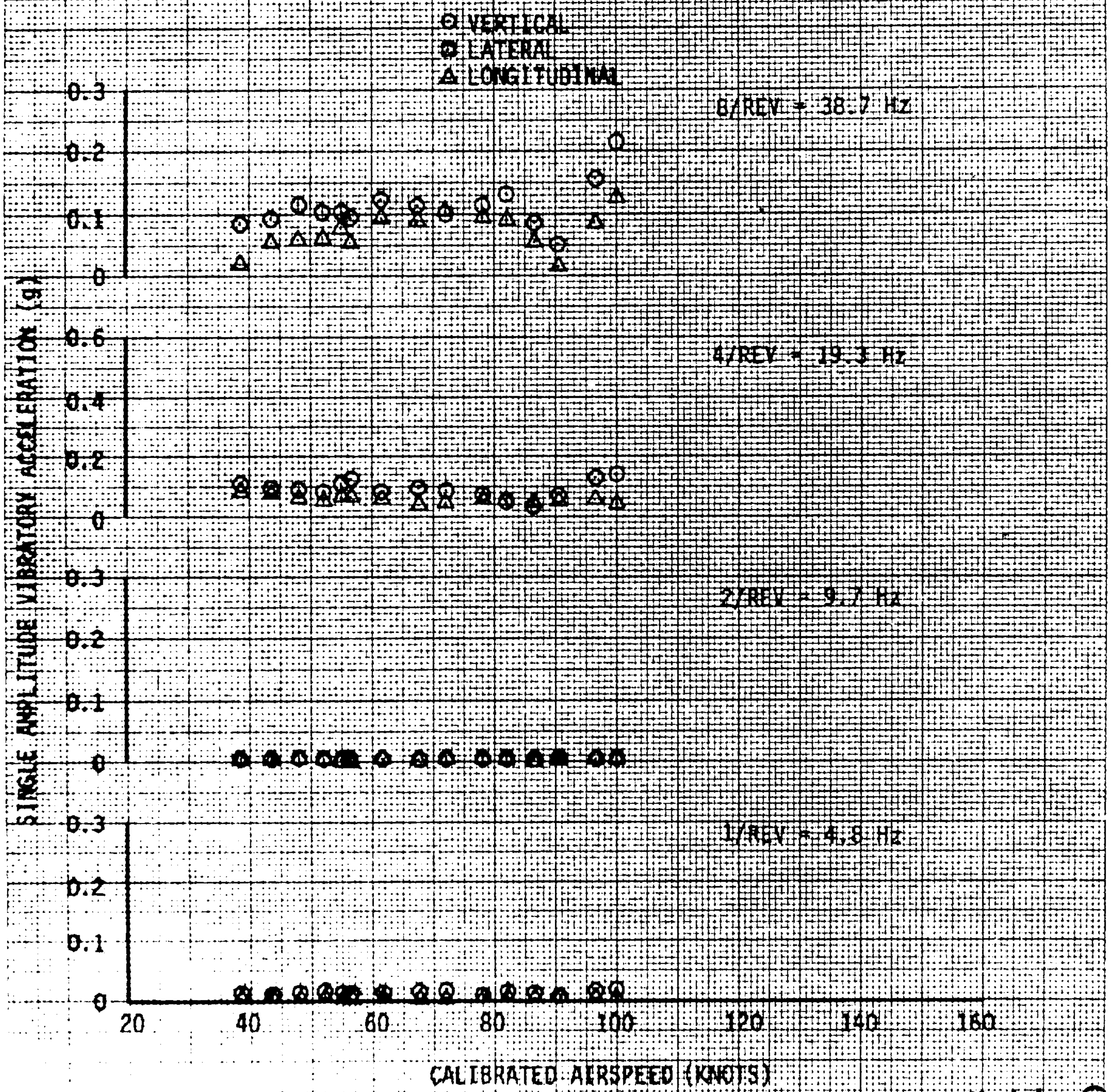

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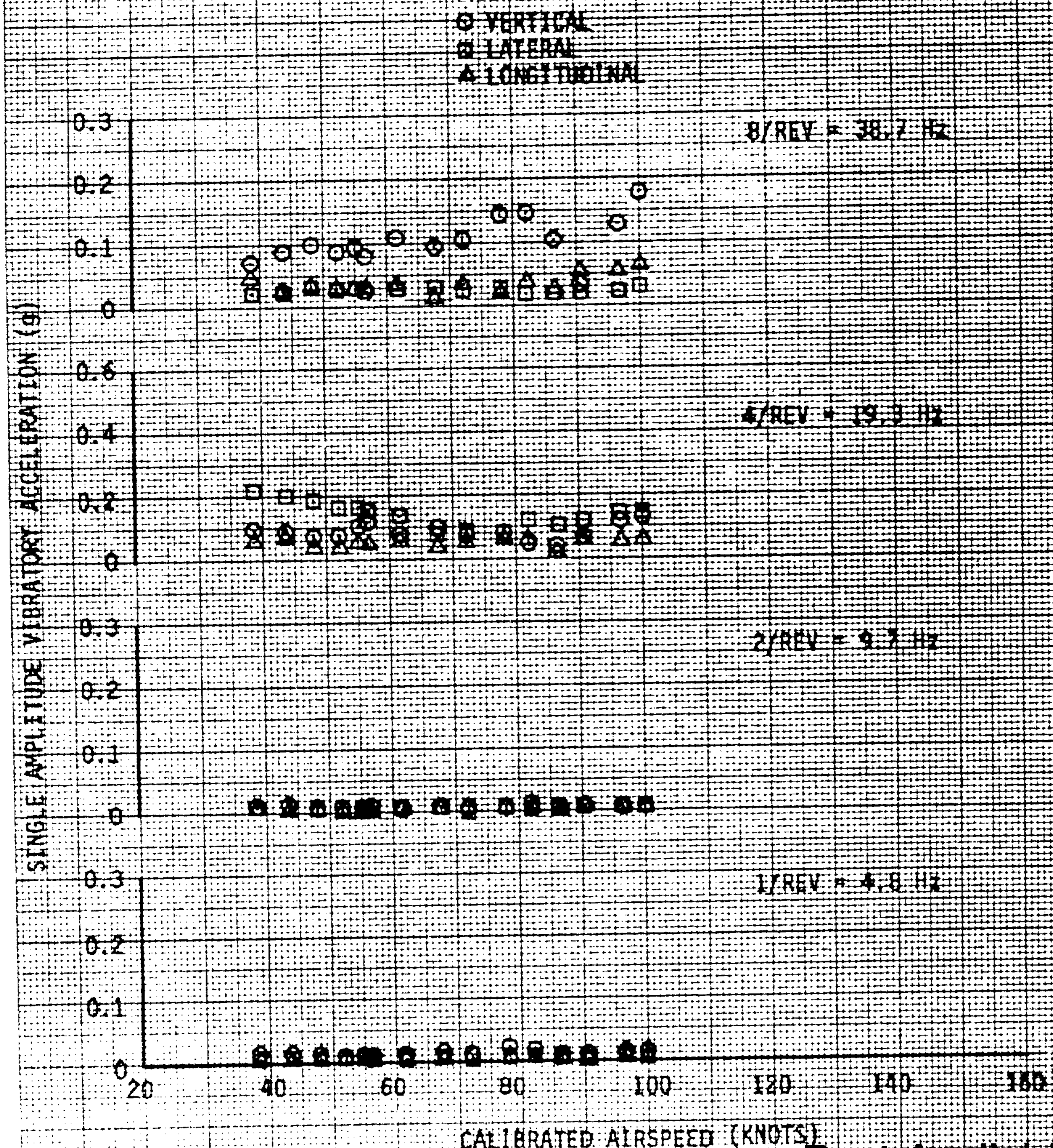
FIGURE 99
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,360	202.1(FWD)	0.8(LT)	15,800	3.5	290	LEVEL

NOTES: 1. PILOT FLOOR
2. 8-HELLFIRE CONFIGURATION



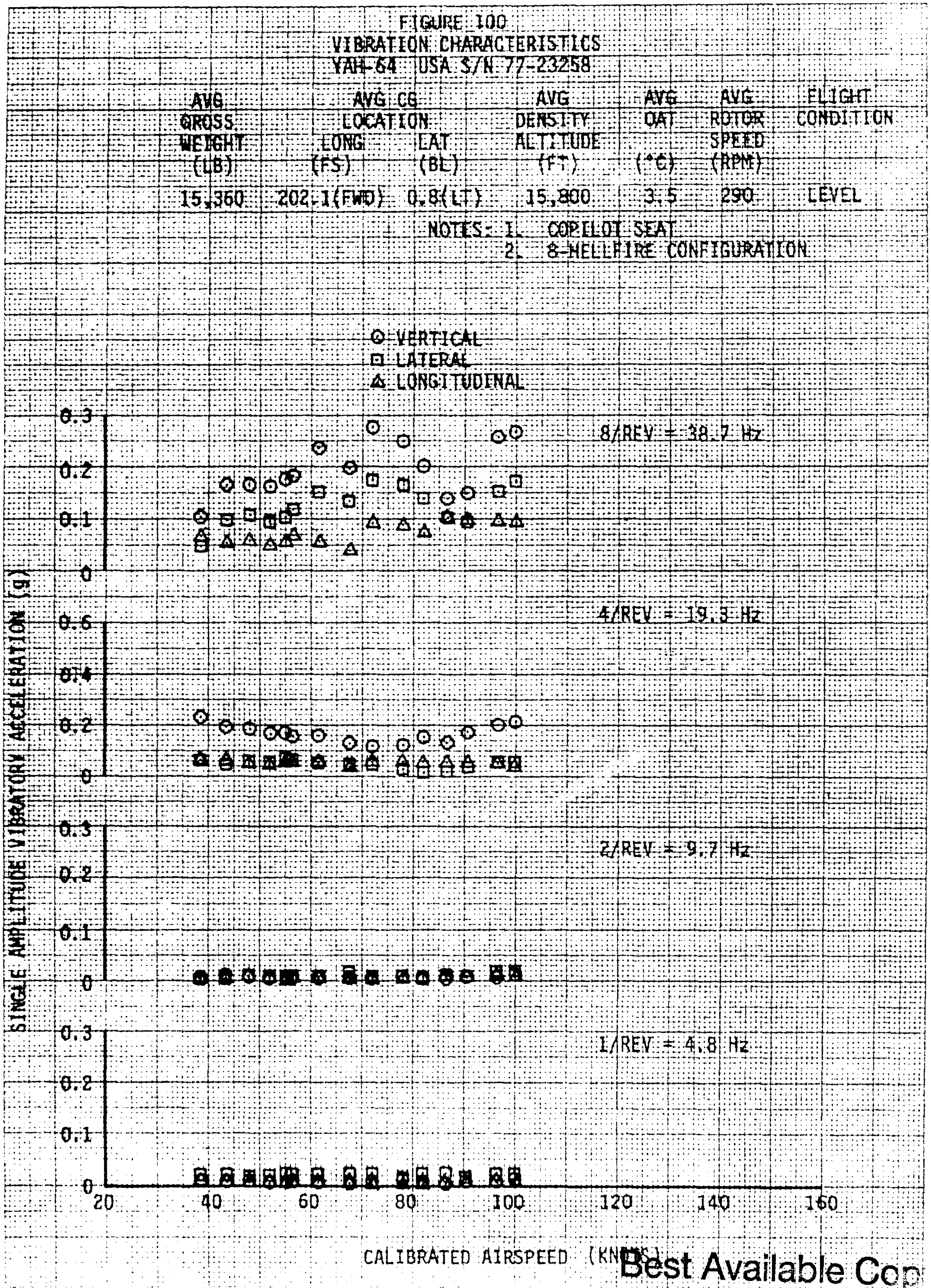


FIGURE 100
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,360
202.1(FWD)
0.8(LT)
15,800
3.5
290
LEVEL
NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION
VERTICAL
LATERAL
LONGITUDINAL
8/REV = 38.7 Hz
4/REV = 19.3 Hz
2/REV = 9.7 Hz
1/REV = 4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
CALIBRATED AIRSPEED (KNOTS)
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FIGURE 101
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,360	202.1(FWD)	0.8(LT)	15,800	3.5	290	LEVEL

NOTES: 1. COPILOT FLOOR
2. 8-HELLFIRE CONFIGURATION

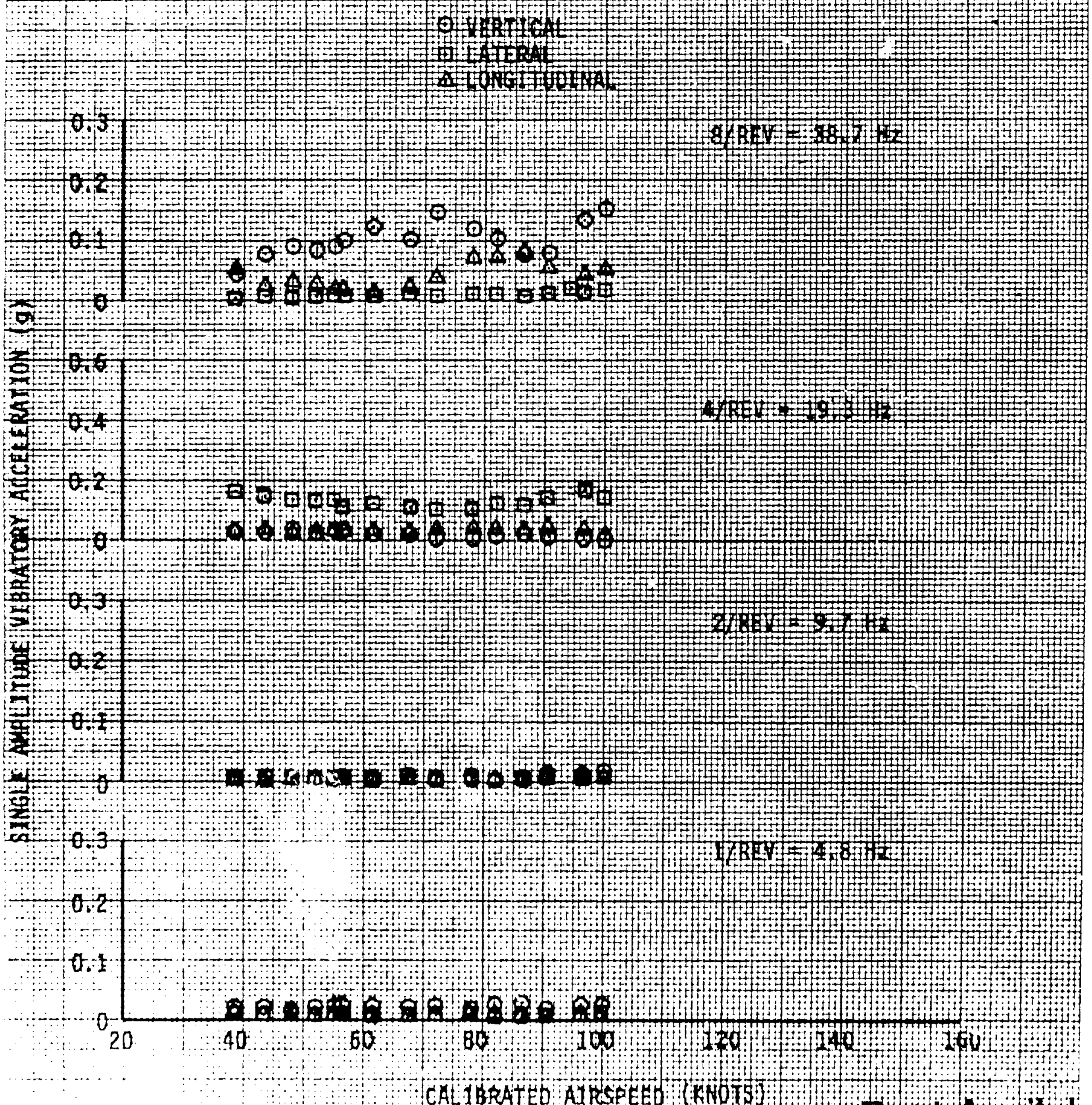

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FIGURE 102
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,360	202.1(FWD)	0.8(LT)	15,800	3.5	290	LEVEL

NOTES: 1. AIRCRAFT CENTER OF GRAVITY
2. 8-HELLFIRE CONFIGURATION

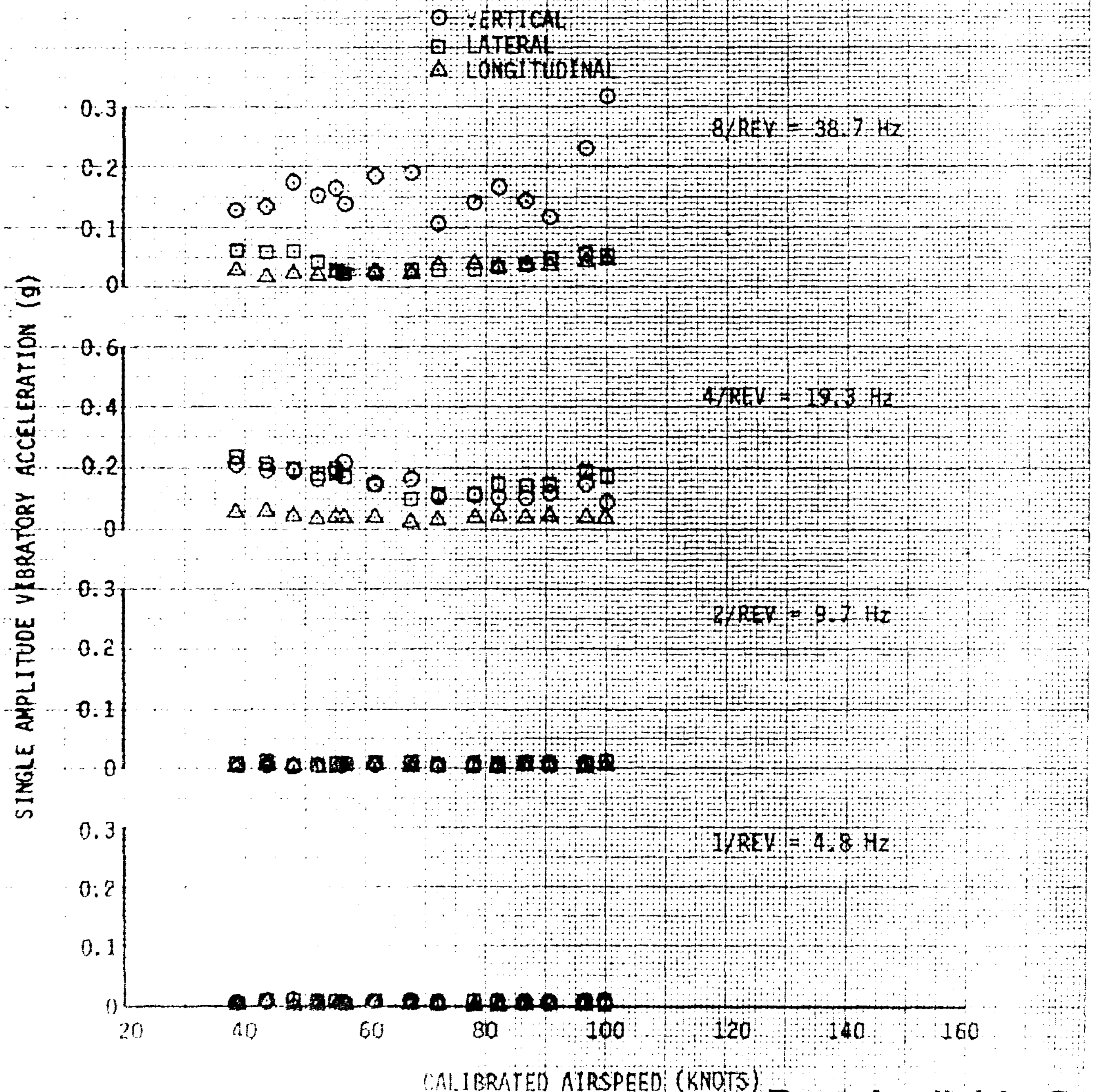

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FIGURE 103
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,700	202.3(FWD)	0.8(LT)	15,320	-0.5	290	LEVEL

NOTES: 1. PILOT SEAT
2. 2-EXTERNAL FERRY TANK CONFIGURATION

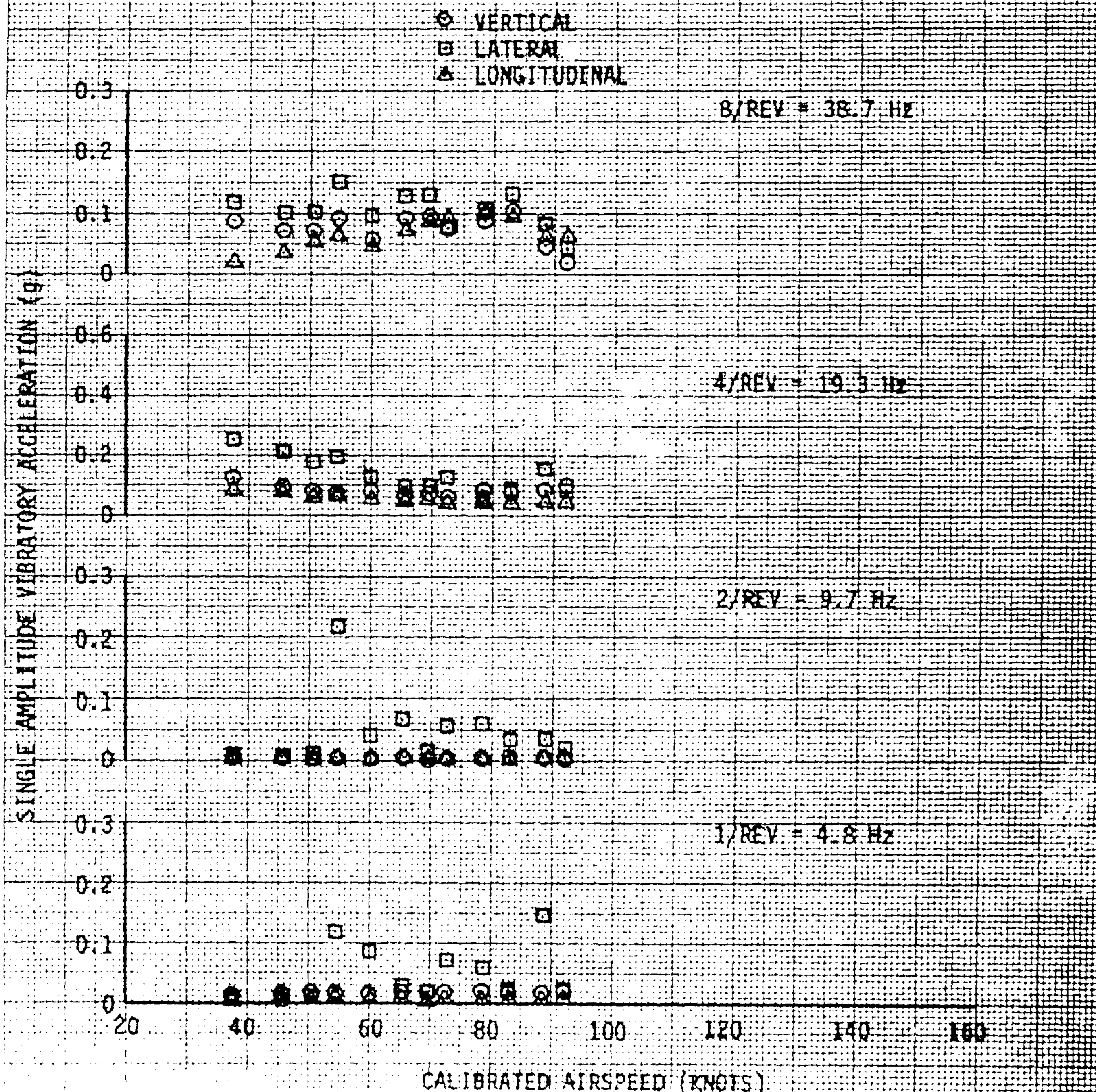

FIGURE 104
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,700	202.3(FWD)	0.8(LT)	15,320	-0.5	290	LEVEL

NOTES: 1. PILOT FLOOR
2. 2-EXTERNAL FERRY TANK CONFIGURATION

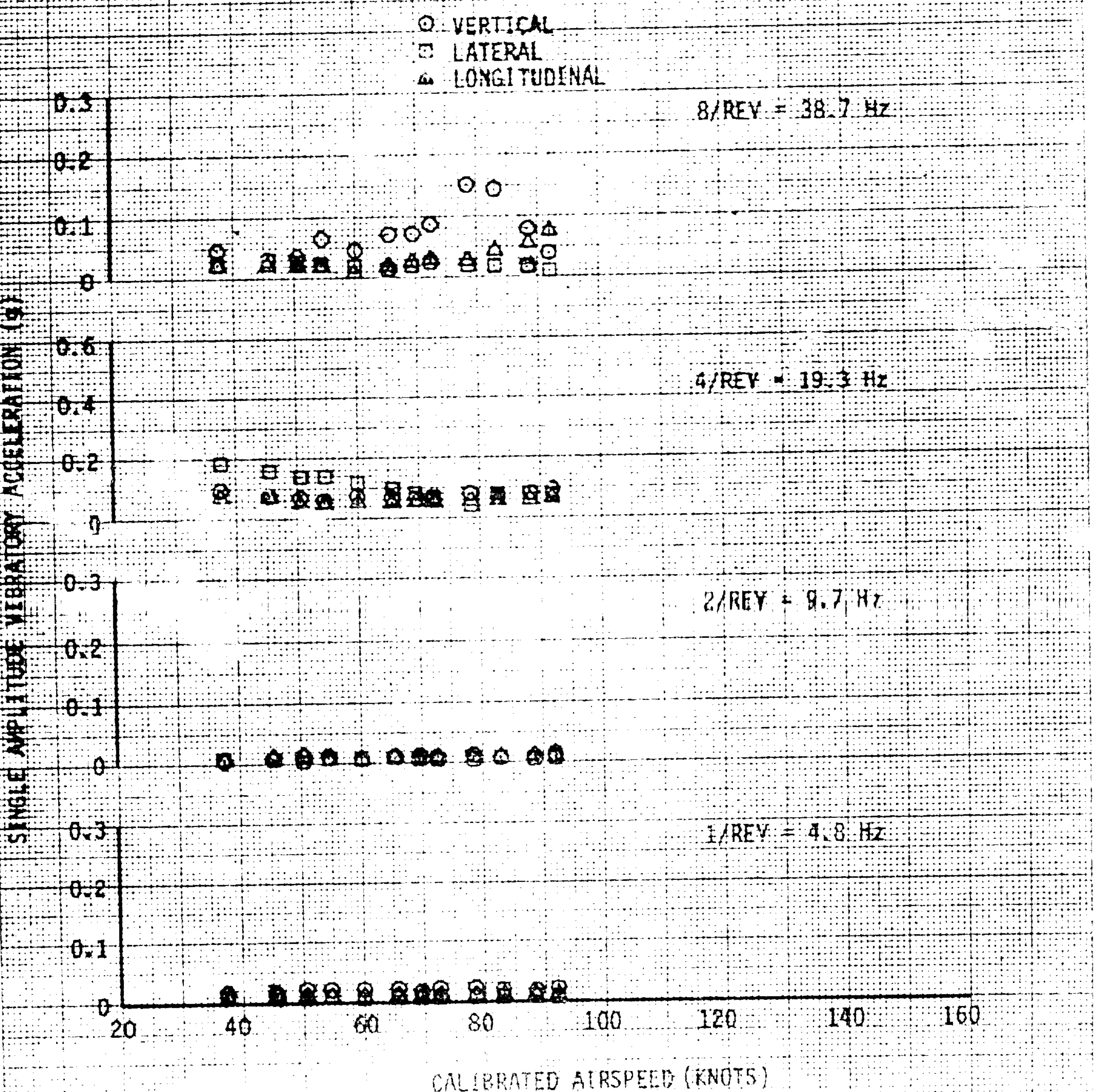

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FIGURE 105
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,700	202.3(FWD)	0.8(LT)	15,320	-0.5	290	LEVEL

NOTES: 1. COPILOT SEAT
2. 2-EXTERNAL FERRY TANK CONFIGURATION

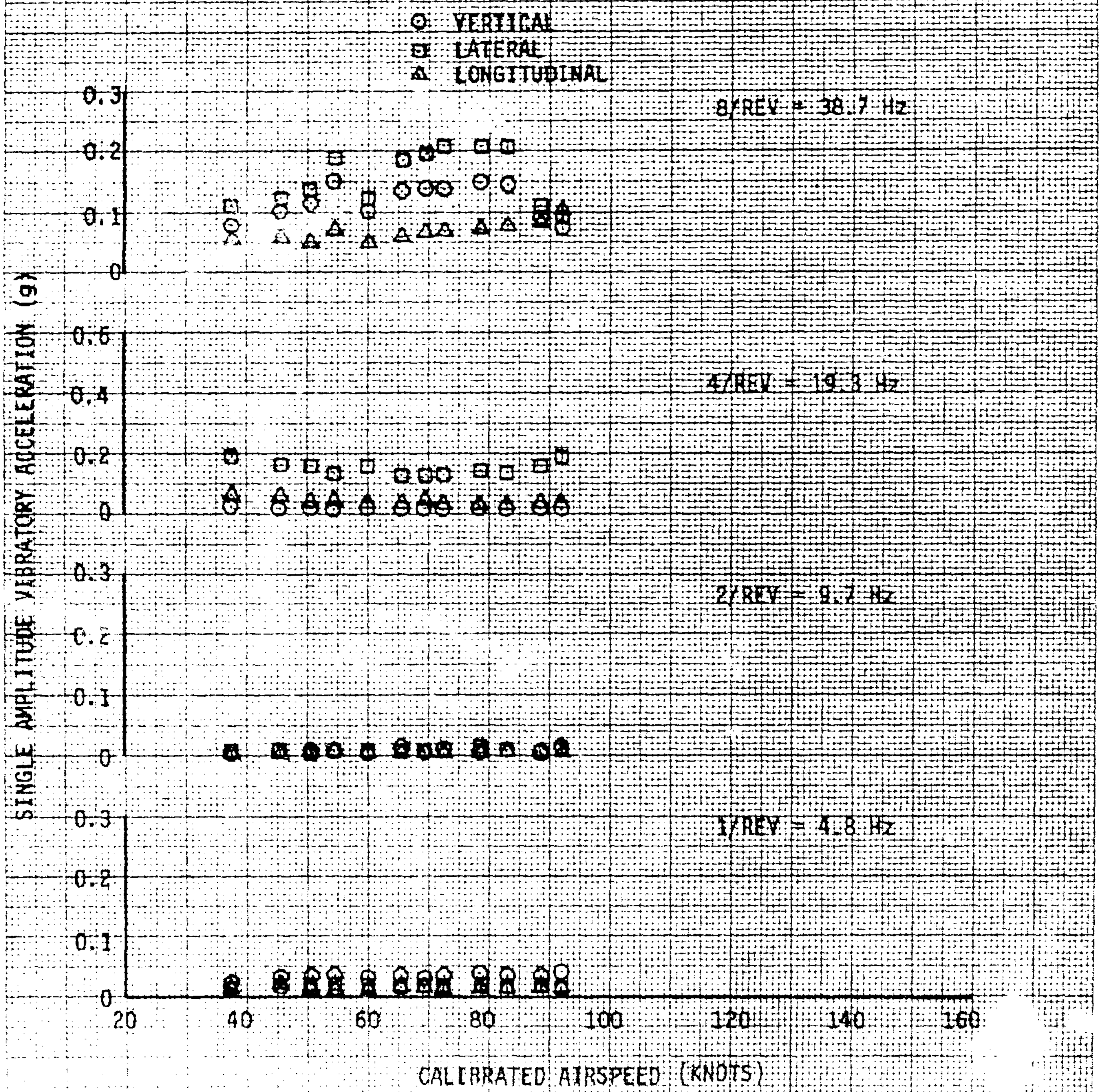

FIGURE 106
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,700	202.3(FWD)	0.8(LT)	15,320	-0.5	290	LEVEL

NOTES: 1. COPILOT FLOOR
2. 2-EXTERNAL FERRY TANK CONFIGURATION

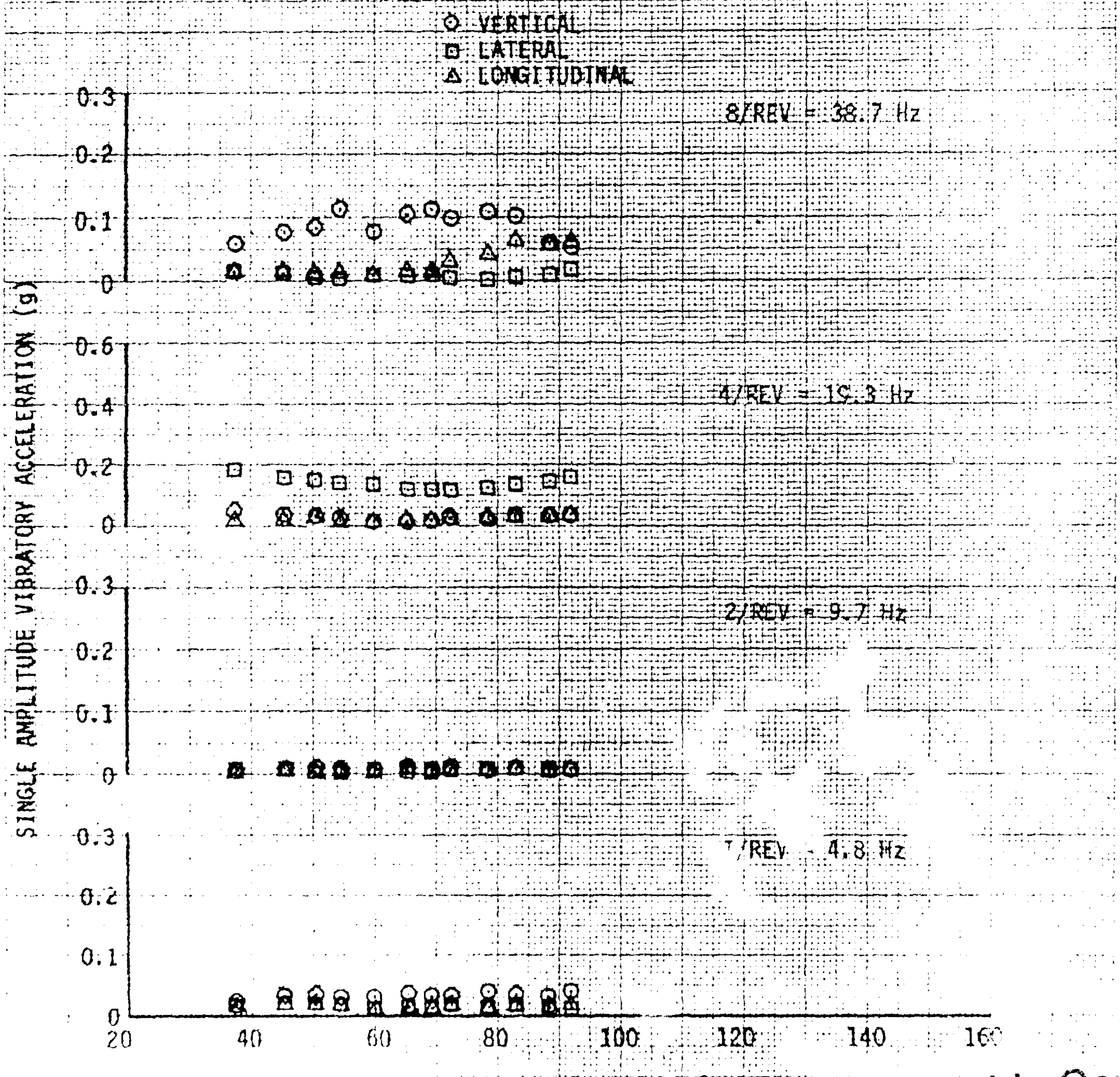

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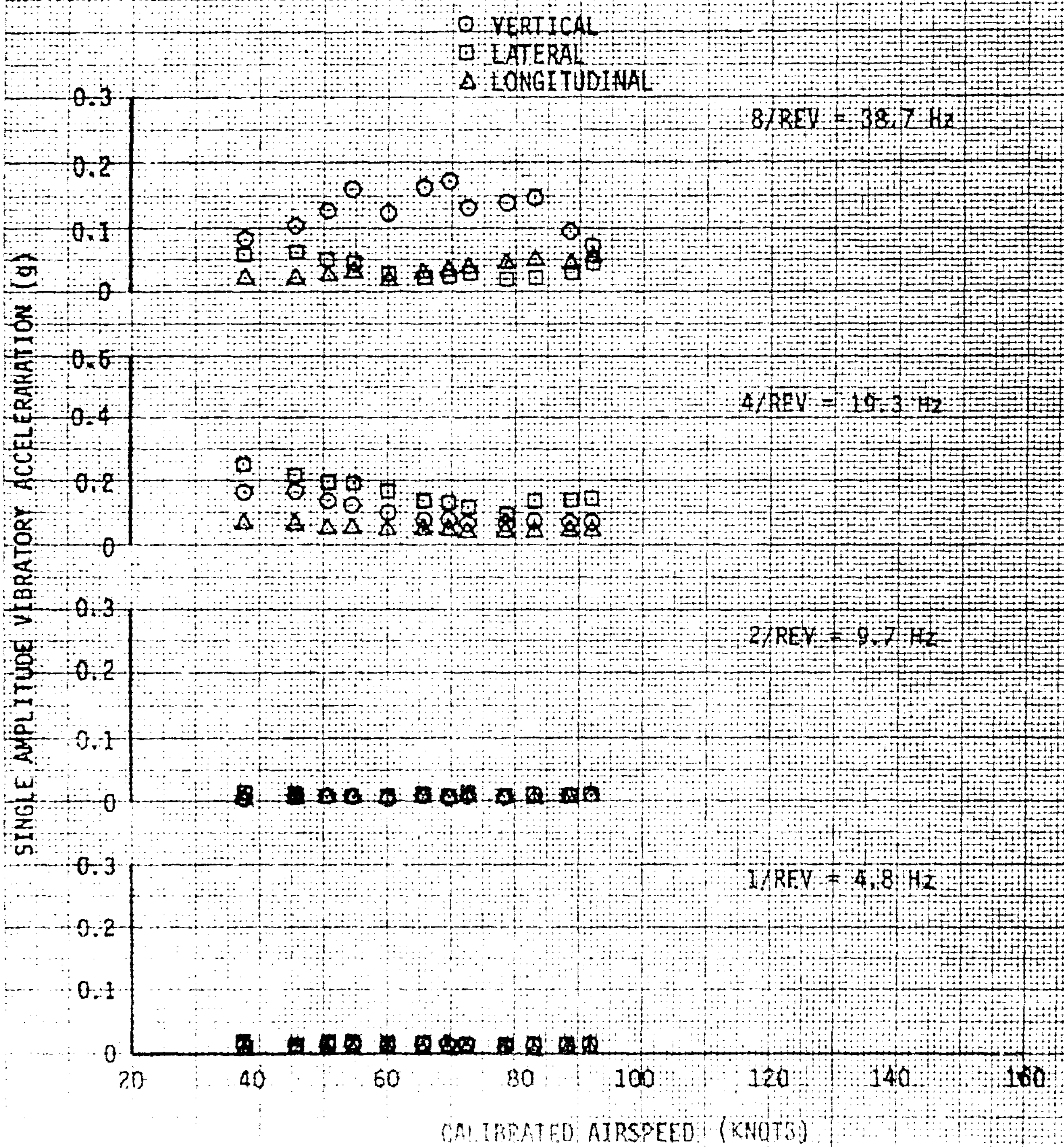
FIGURE 107
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,700	202.3(FWD)	0.8(LT)	15,320	-0.5	290	LEVEL

NOTES: 1. AIRCRAFT CENTER OF GRAVITY
2. 2-EXTERNAL FERRY TANK CONFIGURATION



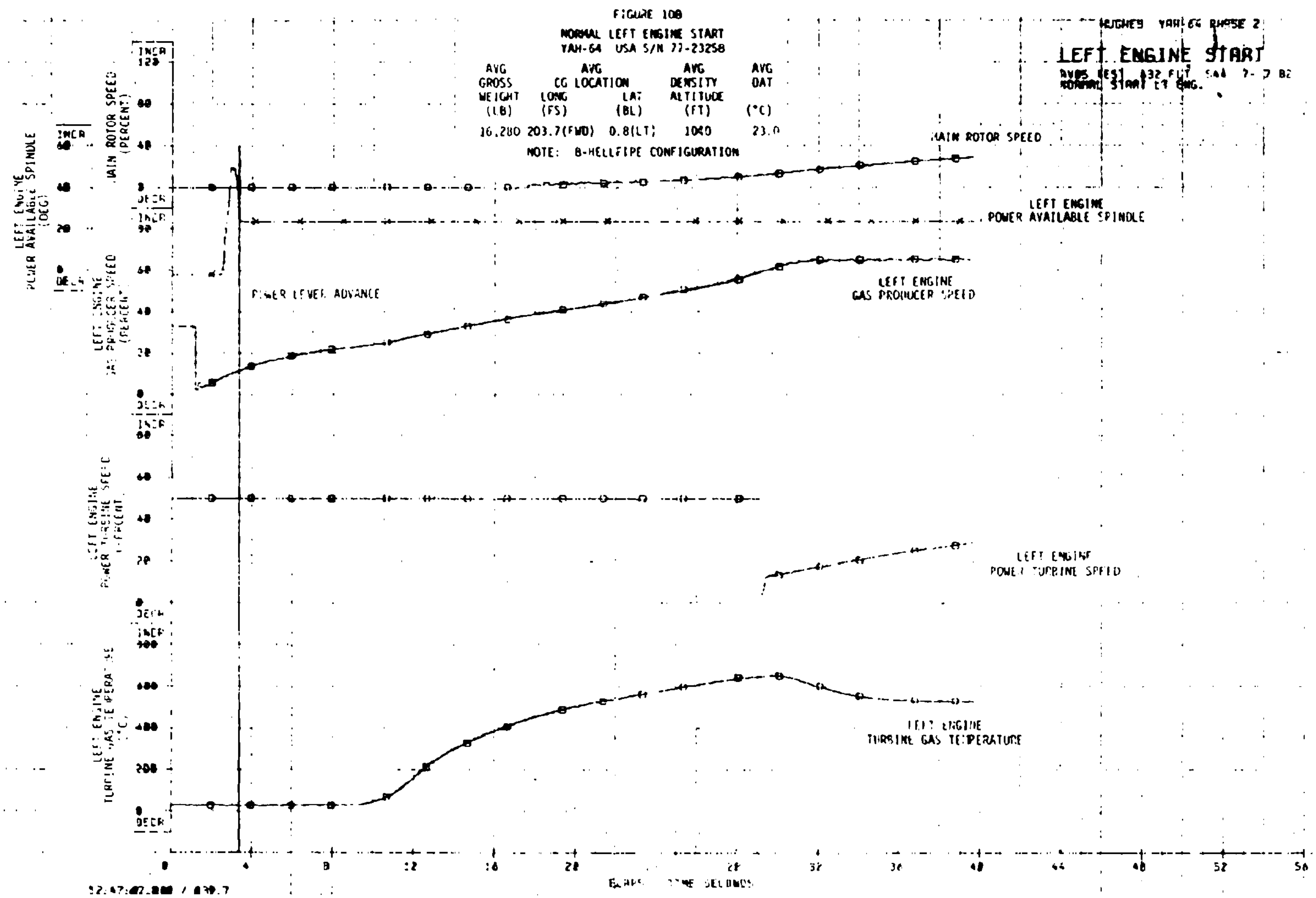


FIGURE 108
NORMAL LEFT ENGINE START
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 16,280
AVG CG LOCATION LONG (FS) 203.7(FWD) LAT (BL) 0.8(LT)
AVG DENSITY ALTITUDE (FT) 1040
AVG OAT (°C) 23.0
NOTE: 8-HELLFIRE CONFIGURATION
HUGHES YAH-64 PHASE 2
LEFT ENGINE START
NORMAL START LT ENG.
MAIN ROTOR SPEED (PERCENT)
LEFT ENGINE POWER AVAILABLE SPINDLE (DEG)
LEFT ENGINE GAS PRODUCER SPEED (PERCENT)
LEFT ENGINE POWER TURBINE SPEED (PERCENT)
LEFT ENGINE TURBINE GAS TEMPERATURE (°C)
INCR
DECR
MAIN ROTOR SPEED
LEFT ENGINE POWER AVAILABLE SPINDLE
POWER LEVER ADVANCE
LEFT ENGINE GAS PRODUCER SPEED
LEFT ENGINE POWER TURBINE SPEED
LEFT ENGINE TURBINE GAS TEMPERATURE
ELAPSED TIME SECONDS
0 4 8 12 16 20 24 28 32 36 40 44 48 52 56

FIGURE 109
NORMAL RIGHT ENGINE START
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)
15,180	201 7(FWD)	0.8(LT)	1880	22.0

NOTE: 8-HELLFIRE CONFIGURATION

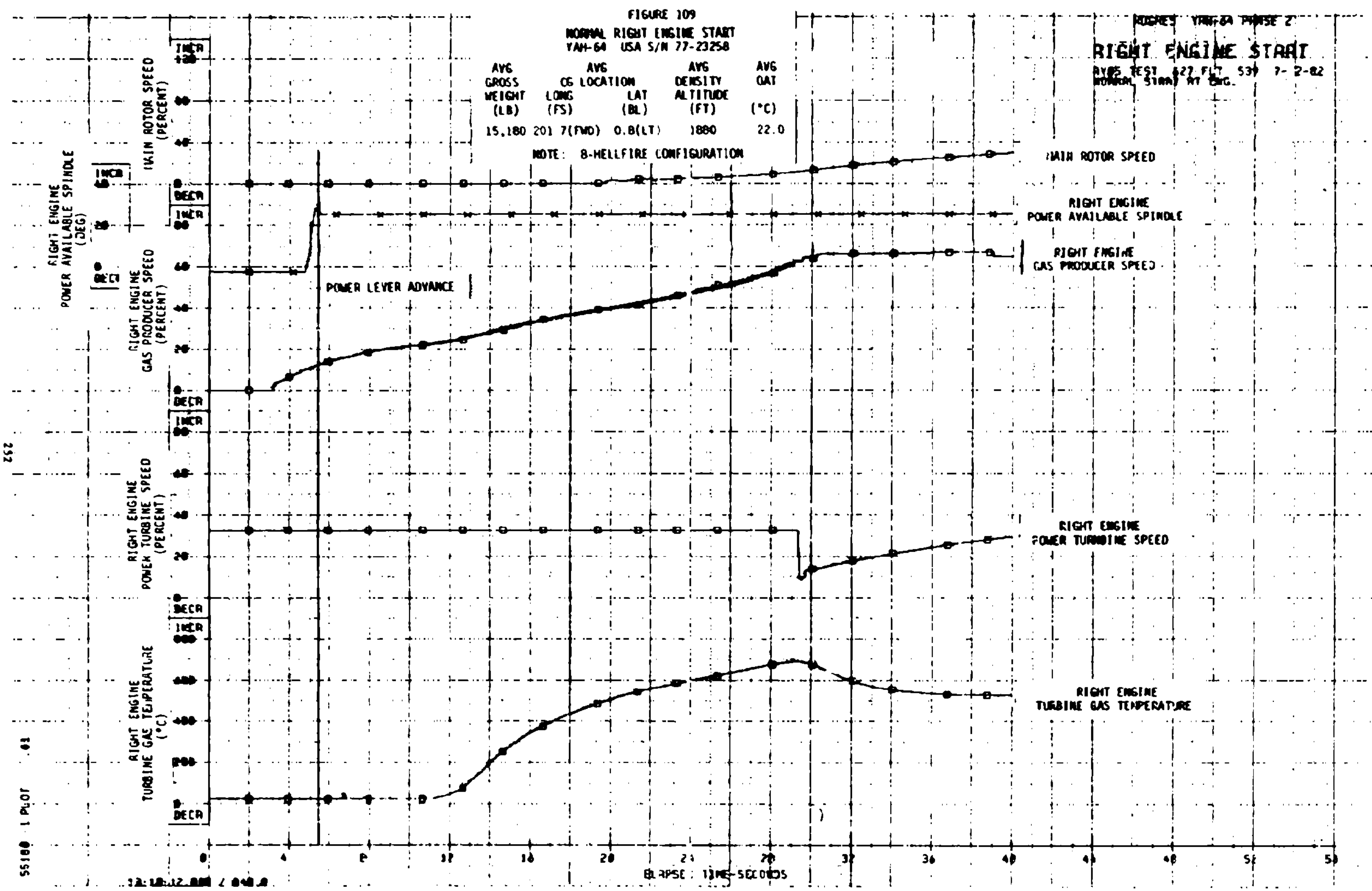


FIGURE 110
NORMAL LEFT ENGINE START
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)
14,360	204.9(MID)	0.8(LT)	5660	20.5

NOTES: 1. CLEAN CONFIGURATION
2. 10-MINUTE HEAT SOAK BACK PERIOD PRIOR TO START
3. MAIN ROTOR SPEED 289 RPM

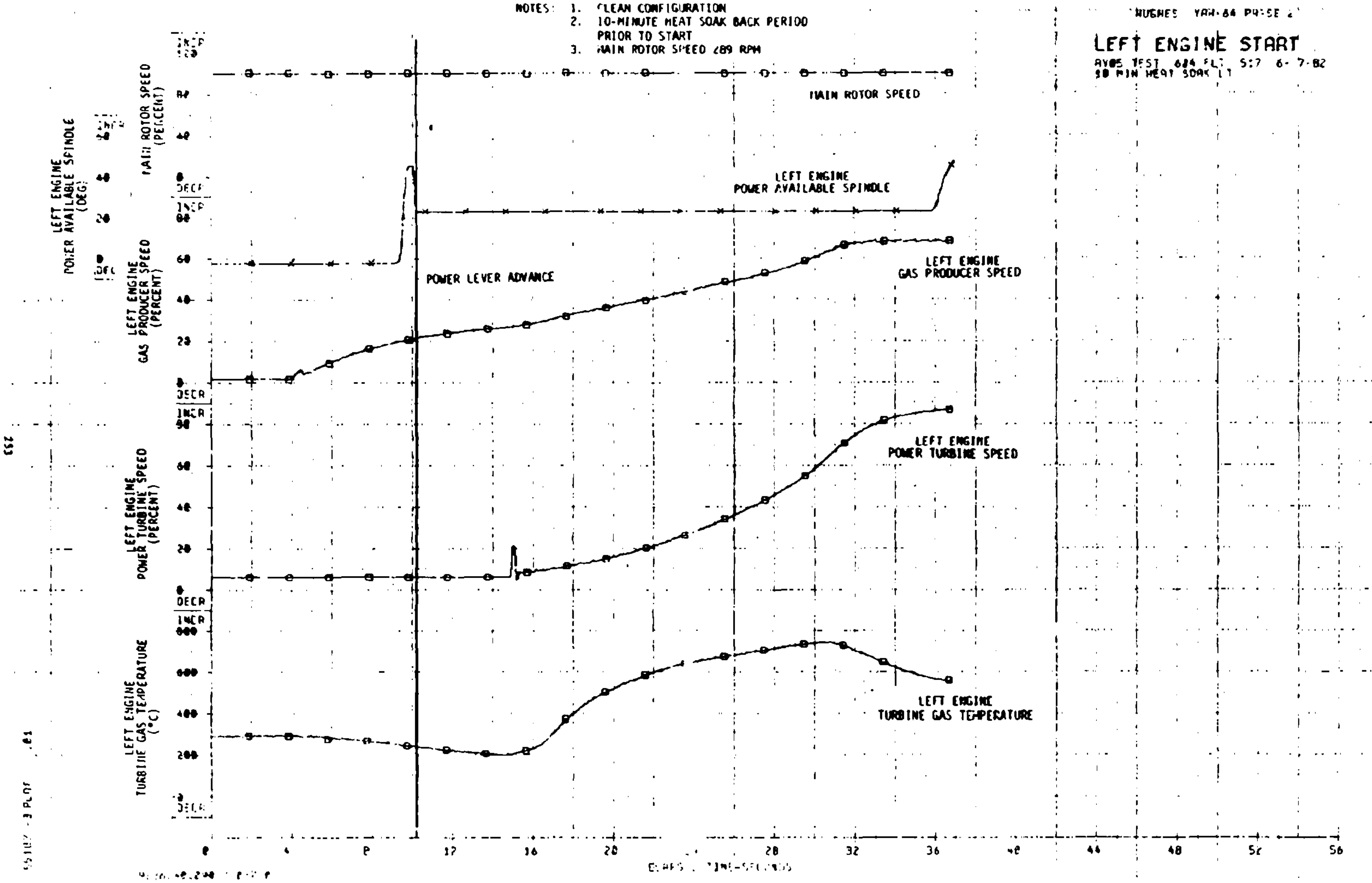


FIGURE 111

NORMAL RIGHT ENGINE START
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)
14,360	204.9(AID)	0.8(LT)	1920	23.0

NOTES: 1. CLEAN CONFIGURATION
2. 10-MINUTE HEAT SOAK BACK PERIOD PRIOR TO START

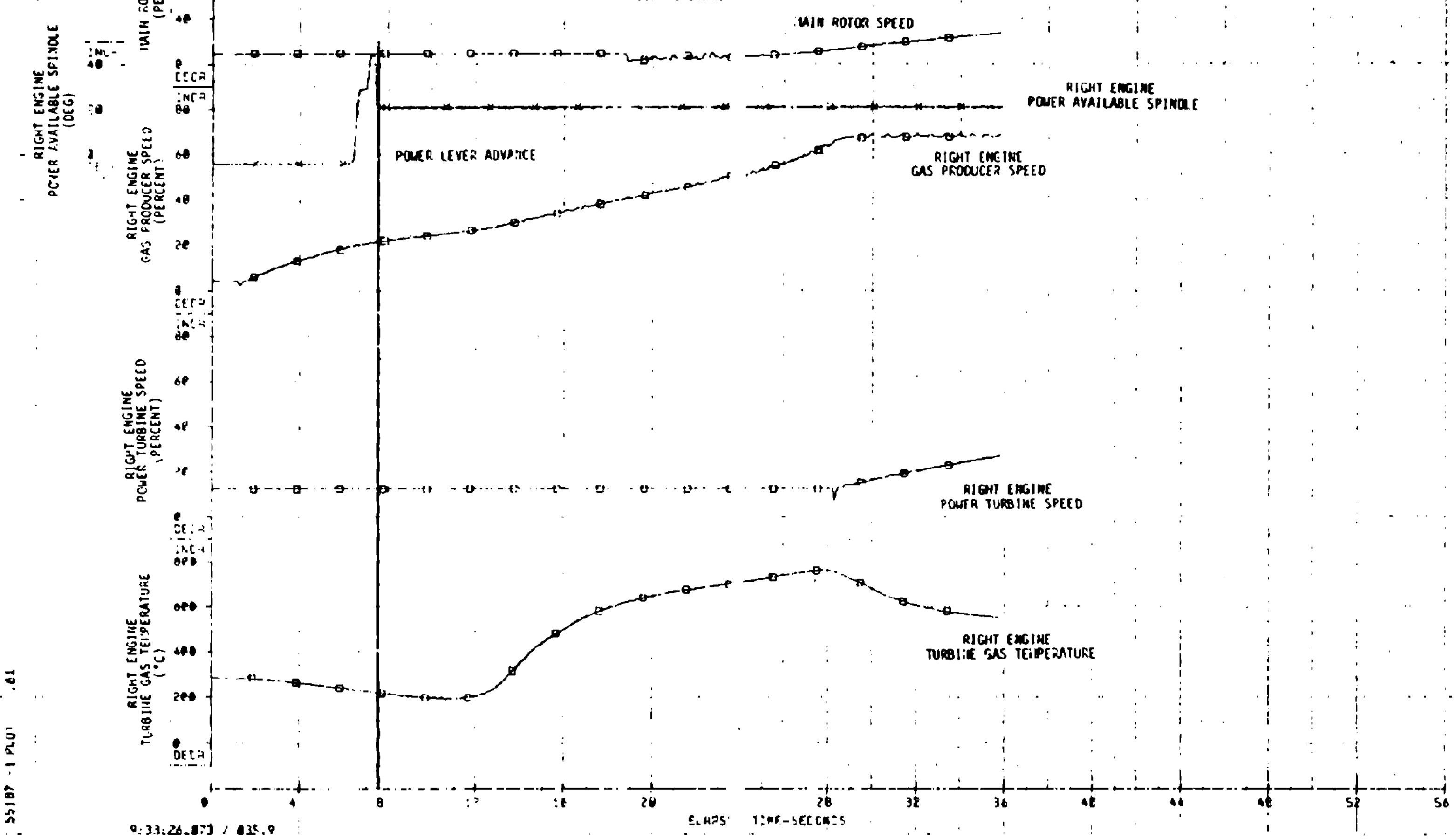
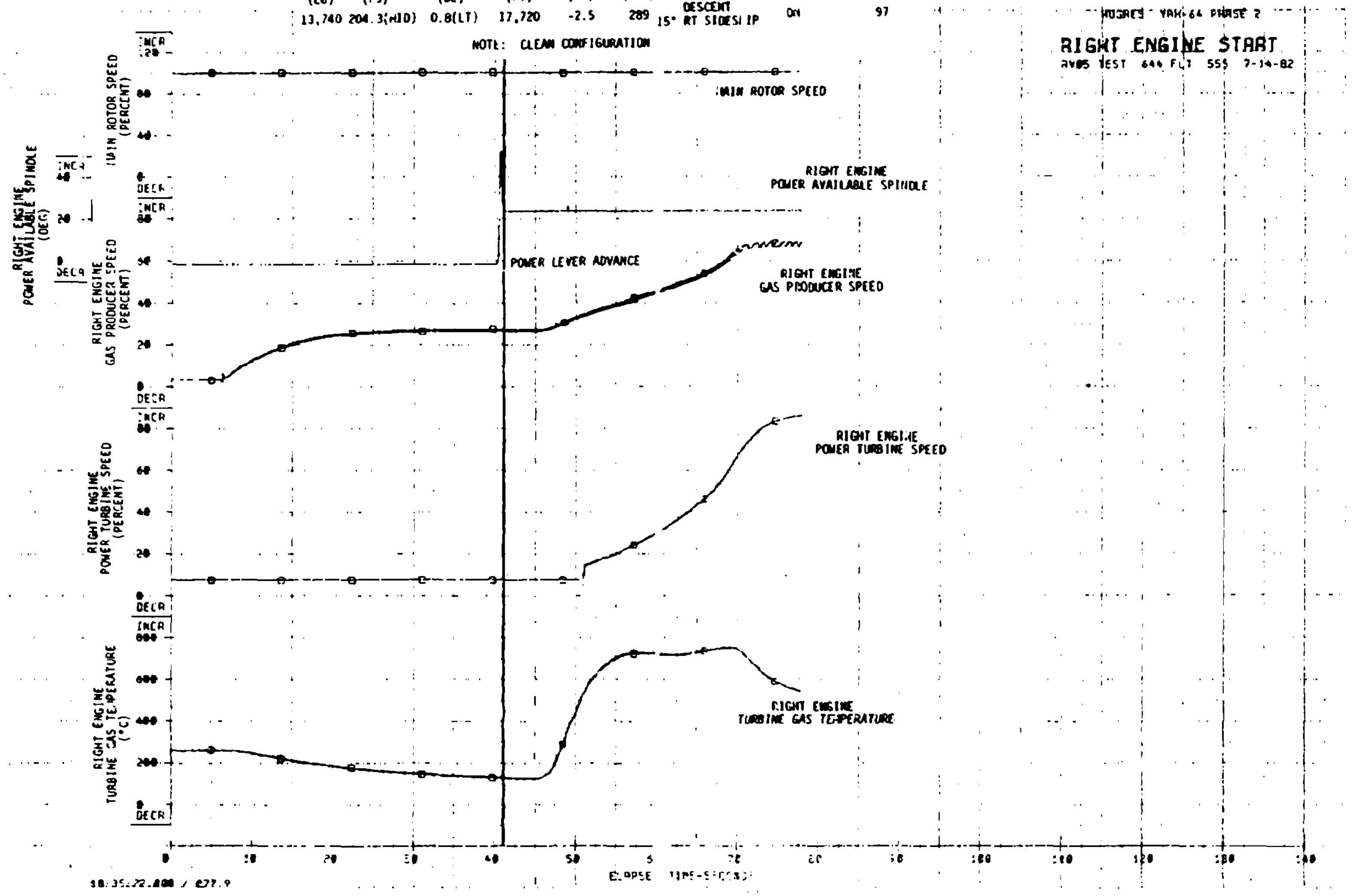


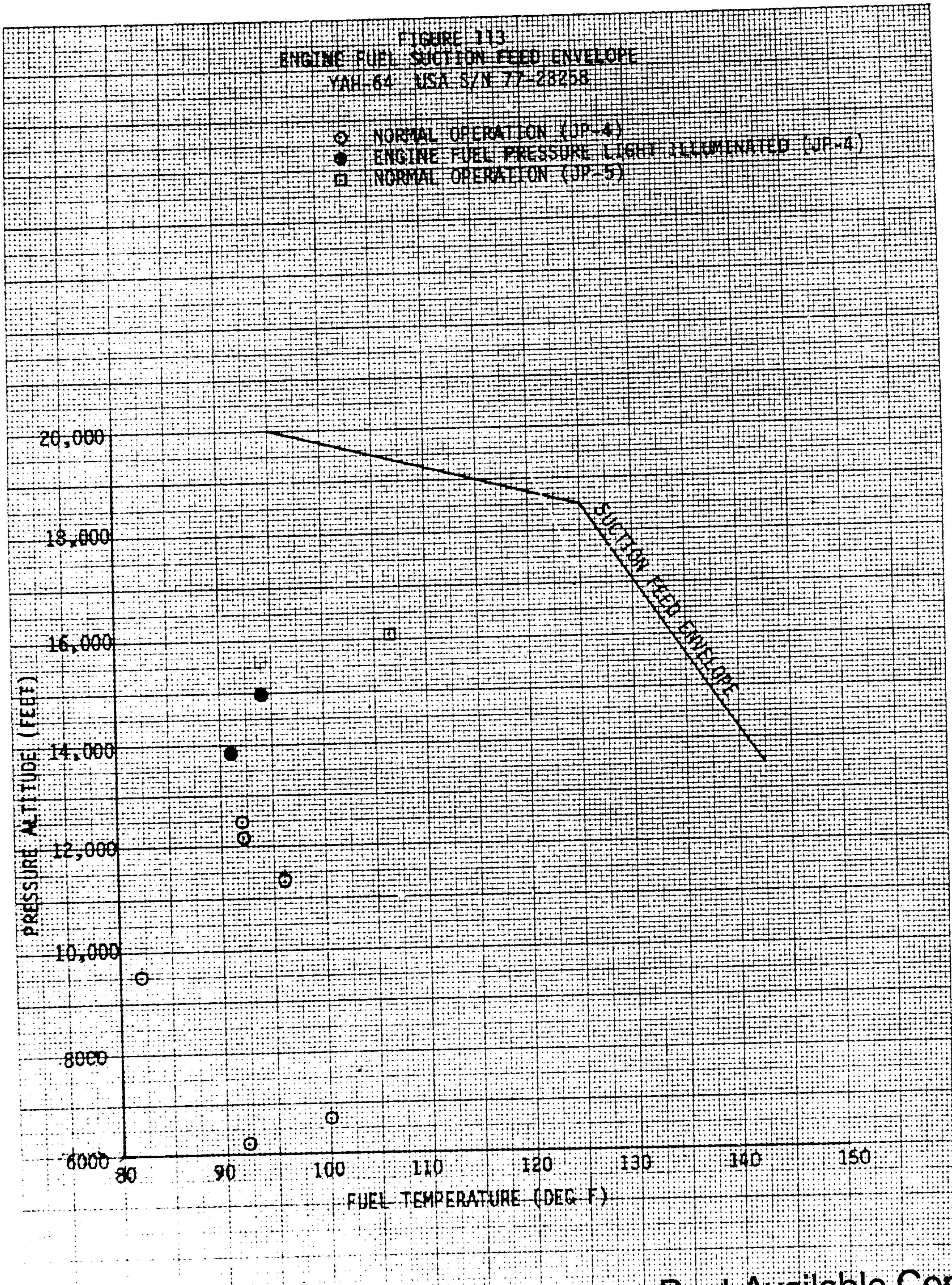
FIGURE 112

INFLIGHT RIGHT ENGINE RESTART
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
13,740	204.3(MID)	0.8(LT)	17,720	-2.5	289	DESCENT 15° RT SIDESLIP	ON	97

NOTE: CLEAN CONFIGURATION





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APPENDIX F. EQUIPMENT PERFORMANCE REPORTS

The following EPRs were submitted.

Number	Subject
80-17-3-1	Pneumatic system check valve broken*
80-17-3-2	Suspected fuel siphoning with APU
80-17-3-3	Heading and Attitude Reference Set alignment
80-17-3-4	Main transmission oil filter buttons popping*
80-17-3-5	Stabilator pivot bushing failures*
80-17-3-6	No. 1 Engine nose gear box chip light*
80-17-3-7	Dummy TADS cover blew off*
80-17-3-8	No. 2 Engines N_g engine out warning activation
80-17-3-9	No. 3 Main rotor blade trailing edge delamination*
80-17-3-10	Trailing edge delamination of all four main rotor blades*
80-17-3-11	No. 2 Engine nose gear box chip light*

*Corrective action taken during test program

APPENDIX G. ABBREVIATIONS

a	Speed of Sound
A	Main Rotor Disc Area (ft^2)
AC	Alternating Current
ADS	Air Data System
app	Appendix
APU	Auxiliary Power Unit
AVRADCOM	US Army Aviation Research and Development Command
A&FC	Airworthiness and Flight Characteristics
BL	Butt Line
BUCS	Back Up Control System
C	Celsius
CAS	Command Augmentation System
cg	Center of Gravity
C_L	Centerline
C_P	Coefficient of Power
CPG	Copilot/Gunner
C_T	Coefficient of Thrust
DASE	Digital Automatic Stabilization Equipment
DASEC	Digital Automatic Stabilization Equipment Computer
DC	Direct Current
deg	Degree
DT	Development Test
EADI	Electronic Attitude Direction Indicator
ECU	Electrical Control Unit
EDT	Engineer Design Test
ENCU	Environment Control Unit
EPR	Equipment Performance Report
ETL	Effective Translational Lift
ETP	Experimental Test Procedure
F	Fahrenheit
f_e	Equivalent Flat Plate Area
fig.	Figure
FS, fs	Fuselage Station
ft	Feet
g	Acceleration of Gravity
GCT	Government Competitive Test
GE	General Electric
GW	Gross Weight
HARS	Heading and Attitude Reference System
HAS	Hover Augmentation System
HHI	Hughes Helicopters Incorporated
HMU	Hydromechanical Unit
H_P	Pressure Altitude
HQRS	Handling Qualities Rating Scale
Hz	Hertz
IGE	In-Ground Effect

IMC	Instrument Meteorological Conditions
in.	Inches
IR	Infrared
IRP	Intermediate Rated Power
KCAS	Knots Calibrated Airspeed
KIAS	Knots Indicated Airspeed
KTAS	Knots True Airspeed
K_P	Power Correction Factor
K_W	Weight Correction Factor
LED	Leading Edge Down
LEU	Leading Edge Up
LB/lb	Pound
LVDT	Linear Variable Differential Transformer
NAMPP	Nautical Air Miles Per Pound of Fuel (specific range)
NOE	Nap of the Earth
N_G	Gas Generator Speed
N_P	Power Turbine Speed
N_R	Main Rotor Speed
OGE	Out-of-Ground Effect
P	Pressure
PCM	Pulse Code Modulation
PNVS	Pilot Night Vision System
PQT-G	Prototype Qualification Test-Government
psia	Pounds per Square Inch, Absolute
psig	Pounds per Square Inch, Gauge
PVT-G	Production Validation Test-Government
Q	Engine Output Shaft Torque
R	Radius (ft)
R/C	Rate of Climb
R/D	Rate of Descent
ref	Reference
RPM	Revolutions Per Minute
SAS	Stability Augmentation System
SCU	Stabilator Control Unit
SDC	Shaft Driven Compressor
sec	Second
SHP, shp	Shaft Horsepower
S/N	Serial Number
T	Temperature
TADS	Target Acquisition and Designation System
TGT	Turbine Gas Temperature
USAAEFA	US Army Aviation Engineering Flight Activity
V_{cal}	Calibrated Airspeed
VDC	Volts Direct Current
V_E	Equivalent Airspeed
V_H	Maximum Horizontal Velocity
V_{ic}	Airspeed Instrument Error Correction

V_{in}	Indicated Airspeed
VMC	Visual Meteorological Conditions
$V_{min\ R/D}$	Optimum Airspeed for Maximum Rate of Descent
V_{NE}	Never Exceed Airspeed
V_{pc}	Airspeed Position Error Correction
V_T	True Airspeed
VRS	Vibration Rating Scale
WL	Water Line
W_f	Fuel Flow Rate

Greek and Miscellaneous Symbols

Δ	Incremental Change
μ	Advance Ratio
ρ	Air Density (slugs/ft^3)
σ	Air Density Ratio
Ω	Main Rotor Angular Velocity (radians/sec)
1/rev	1st Harmonic of the Main Rotor
2/rev	2nd Harmonic of the Main Rotor
4/rev	4th Harmonic of the Main Rotor
8/rev	8th Harmonic of the Main Rotor
ψ	Main Rotor Blade Azimuth Position

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US Army Aviation Center (ATZQ-D-T, ATZQ-TSM-A, ATZQ-TSM-S, ATZQ-TSM-U)	4
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US Army Research and Technology Laboratories (DAVDL-AS, DAVDL-POM (Library))	2
MTMC-TEA (MTT-TRC)	1
ASD/AFXT	1
Advanced Attack Helicopter Program Manager (DRCPM-AAH-SE, DRCPM-AAH-APM-TE)	6
US Army Operational Test and Evaluation Agency (CSTE-TM-AV)	1
General Electric - AEG (Mr. Koon)	1
US Army Materiel System Analysis Agency (DRXSY-AAS)	1
Hughes Helicopters (Gerry Ryan)	3
US Army Missile Command (DRCPM-HDT-T)	1
Foreign Science Technology Center (DRXST-BA)	1